LUNAR ORBITER CAPSULE PRELIMINARY CONFIGURATION REPORT

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FOREWORD

The first goal of the Lunar Orbiter Capsule Study, Phase II, was the selection of a specific capsule configuration for detailed analysis. The various trade-offs to be considered were initially studied during Phase I of this contract, and the effects of these trade-offs have been evaluated in arriving at a recommended system. In some areas it was necessary to extend the results of the Phase I study in order to carry out an effective comparison among the various system choices. This has been done primarily in the areas of systems analysis power supply requirements, and attitude control and stabilization. The general mission objectives and design criteria to be used in governing the various system choices were delineated during this period and are given in this report. The results of this initial effort of the study are presented herein together with recommendations for a specific configuration of the Lunar Orbiter Capsule for the approval of the Jet Propulsion Laboratory.

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SECTION I

SUMMARY OF RECOMMENDED CONFIGURATION

A. MISSION OBJECTIVES AND DESIGN CRITERIA

1. MISSION OBJECTIVE

The primary objective of the Lunar Orbiter Capsule (LOC) is the procurement of television pictures of the lunar surface with resolution and coverage sufficient to delineate those lunar surface characteristics which are important to the Apollo mission. Specifically, the mission should be capable of surveying various areas of the lunar surface for relative roughness so that selection of a landing site may be made and the difficulty of performing the landing of a manned vehicle within the selected area ascertained. It is anticipated that one or more orbiting payloads may be flown within the flight schedules for Rangers 10 through 14. As a secondary mission objective, a tracking transponder similar to those presently planned for use on the Ranger Spacecraft may be included to determine the selenodetic gravitational potential constants. Additional scientific experiments may be flown whenever sufficient payload and other design criteria will permit such inclusions. However, each of the additional experiments must be included on a non-interference basis and must not divert attention from the primary objective, i.e., the obtaining of TV pictures of the lunar surface.

2. DESIGN CRITERIA

a. Spacecraft Bus Criteria

Considerations of the bus design criteria are presented here for technical completion only. No further considerations of bus design are given in this report.

It will be a design goal that changes in the Ranger 6, 7, 8, and 9 bus configuration will be made only as required to accommodate the LOC with a simple interface (in order to permit the incorporation of non-interfering experiments), or to incorporate mandatory engineering change orders resulting from Ranger 3 through 9 experience. In the event that changes in the bus configuration beyond the establishment of a simple interface are required, the changes will be minimized and will require full approval of JPL. Each change will be evaluated in terms of its effect on the bus and the alternatives available in the design of the LOC without inclusion of the change.

The requirements of the bus will be to deliver the LOC to the vicinity of the moon on a nominal 66-hour transit trajectory within the accuracy limits obtainable with a single mid-course maneuver. The necessary ground-commanded terminal maneuver to orient the LOC prior to separation and injection into lunar orbit will be required of the bus within the tolerances of the Ranger 6 through 9 attitude control and command systems.

- Competing Characteristics

  Competing characteristics in the spacecraft bus criteria will be treated in the following order:

  1. Reliability of Subsystems in the following order of importance:

     (1) Terminal-phase orientation and attitude control

     (2) Injection point determination

     (3) Bus functions prior to terminal phase (separation, stabilization, midcourse maneuver, etc.) as checked out in previous flights.

     (4) Bus failure-detection telemetry

     (5) Passenger experiments and associated data handling systems

  2. Compatibility of LOC and Bus.

  3. Schedule

  4. Suitability of designs and documentation for continued exploitation of Ranger.

     (6) Cost

     (7) Biological Sterility

- Defined Characteristics

1. Weight - The gross weight of the spacecraft is not listed above as a competing characteristic because its value will be fixed by available vehicle performance and the trajectory chosen. The desired approach to the weight problem is, first, to try to minimize the weight of each component and then, second, to abandon plans to fly some components if necessary to reduce the total weight. If the spacecraft with all desired components is underweight, the performance excess will normally be manifested in an altered trajectory or increased propellant residuals in order to obtain higher reliability and/or greater accuracy of injection. Ballast will

normally not be used except when required for balancing the spacecraft. The weight allowance for the LOC has been assumed to be 450 pounds. This assumes an 825-pound injected weight and a 375-pound bus.

2. Effect on Agena B Stage - It is a general spacecraft design criterion that any changes required on the Agena B relative to a standard, contemporary Air Force vehicle shall be held to the absolute minimum necessary to achieve the mission. Reliability, cost, and schedule assurance are all involved in this decision; it is, therefore, a most important constraint.

- Experimental Philosophy Affecting Design

1. Environmental Control Prior to Final Mating of Vehicle with Launcher - The spacecraft and its components shall not be subjected to environments beyond flight-acceptance levels, and the burden of protection shall be on operators and facilities rather than on the spacecraft itself. Clean, air-conditioned working areas, special handling fixtures, air transport, electrical overload protection, and other such measures external to the spacecraft shall be used to the extent practical, and the spacecraft design shall be based primarily on environments expected from the time of final mating onward through launch and flight. Biological sterility shall be achieved by external measures where possible, so that the adverse effects of the sterility requirement on equipment design, reliability and schedule are minimized.

2. Operating Condition at Launch - A list of contingencies will be prepared under which decisions can be reached, in advance of the event, as to policy on holding, scrubbing, or launching with known inoperative components. In general, the design criterion should be that successful flight depends mainly on functional reliability of the subsystems from the end of hangar checkout onward, rather than on instrumentation, control, and correction of deficiencies in the spacecraft after mating with the launch vehicle. Atlas-Agena B operations require that the spacecraft should function unattended for many hours before a shoot, and also for an hour or more after a late hold or scrub. During this period of serveral hours instrumentation beyond normal telemetry and essential power-system controls is undesirable. Operating procedures will restrict on-pad testing to tests of a system-verification nature. Subsystems must be designed to hold calibration and must be completely checked and calibrated before leaving the hangar.

3. In-flight Failure Detection - In keeping with the main objectives of these firings, failure detection on the basic functional systems is desirable. However, since the number of modes of failure may exceed the number of measurements available, an artful programming of the failure-detection system may be required to obtain unambiguous, reliable data (preferably from redundant sources), on vital elements without losing entirely the chance of detecting failures in less important (and possibly less reliable) elements.

4. Function After In-flight Failure - After a successful injection and separation of the spacecraft, omnidirectional tracking capability is required (within practical limits).

Every subsystem in the spacecraft depends on others for its function and, therefore, the question of flight function after failure arises. The general design criterion is that no subsystem is required to function after a failure of the subsystem on which it depends, except insofar as continued function can be achieved without complicating the design or diverting effort from the main objectives of the flight.

In the event that the spacecraft is injected in a functioning configuration but not on a trajectory that will take it to the Moon, the experiment becomes another long-life, long-range test of the system and, therefore, the attitude-control, power, and communication subsystems should be designed for this condition as far as is practical without compromising the primary lunar objective.

Design provisions should permit separation and operation of the LOC system within the system capability for an engineering test, regardless of the trajectory attained.

b. Lunar Orbiter Capsule System Criteria

Simplicity and reliability will be the governing criteria for the LOC within the capabilities necessary to obtain the aforementioned mission objectives. The LOC will be required to provide the necessary retro-propulsion for achieving a lunar orbit of nominal characteristics commensurate with the TV experiment, and to orient and stabilize the vehicle to the lunar local vertical within the tolerances required and for the necessary lifetime associated with the television experiment and the primary mission objective. The power, communications and command control capability will be supplied as necessary to accomplish the mission.

Inclusion of additional experiments to accomplish other than the primary objectives will be made on a non-interference basis.

- Competing Characteristics

Competing chracteristics in the LOC system criteria will be treated in the following order:

1. Reliability of Subsystems in the following order of importance:

   (1) Retro-propulsion capability

   (2) Attitude control and stabilization

   (3) Video transmission link

   (4) Television subsystem

(5) Transponder subsystem

(6) Prognostic and diagnostic telemetry

2. Coverage

3. Limiting Resolution

4. Schedule

5. Cost

6. Biological Sterility

3. TEST PHILOSOPHY AFFECTING DESIGN

In all areas where the design is influenced by the type of test program planned, the design criterion will be that first preference will be given to designs that can be analyzed, and that testing will be resorted to primarily for the purpose of confirming analysis or resolving system interactions. When subsystem qualification tests are the only way of determining readiness for assembly and flight, the design must of course incorporate the necessary test provisions. In general, however, the objective is to obtain reliability by design rather than by testing. In system tests, the primary purpose is to simulate flight operation. The burden of simulation shall, as far as possible, fall on equipment external to the spacecraft.

B. GENERAL SYSTEM PARAMETERS

1. TRANSIT TRAJECTORY AND LAUNCH WINDOW

It is desired to continue the use of the existing Ranger spacecraft design essentially unchanged for the proposed Lunar Orbiter missions. It has been determined that the 66-hour transit trajectory family, as established for the Ranger flights 6 to 9, can be used almost exactly as previously planned. The spacecraft geometry imposes a basic launch constraint in that the sun-probe-earth angle must lie between 45 and 135 degrees for the last 50 to 55 hours of the flight. This constraint permits launch during two 8-day "windows," centered about the first and third quarters, during each synodic month.

The sequence of surface areas illuminated by the sun differs for injections into lunar orbit at first quarter and at third quarter. For first-quarter injections, the trailing-edge hemisphere is first illuminated, then the visible face, then the leading-edge hemisphere, and then the invisible or rear face. The sequence is identical and cyclical for third-quarter injection, except that the process starts with illumination of the leading-edge hemisphere and ends with the visible face.

Since pictures of selected portions of the visible face are the primary mission objective, it would appear reasonable to begin the mapping with an injection near the first quarter in view of the finite probability of failure of some vital part of the picture retrieval system in any time interval. If a truly long-life system were available, little advantage would be obtained in selecting one launch window in preference to the other. In the presence of a relatively short mission lifetime, the most desired functions should be accomplished first, and this means a launch at or near the first quarter, the exact extent of the launch window to be determined later as a function of injection parameters. This does not, however, make it less desirable to plan the mission to make optimum use of the camera system, in that pictures of the rear face of the moon will be available and very much desired; also, a system lifetime near 35 days will permit essentially complete mapping of the visible face of the moon a second time. "Complete mapping" should be understood to mean: available for mapping within the bounds of orbit inclination, illumination, and recording time per orbit.

Midcourse guidance is expected to be applied once, as for the Ranger 6 to 9 missions. Miss-distance dispersions are expected to be as described in JPL TM 312-177[1], wherein the 3-sigma dispersion in radius of closest approach (R. C. A.) is given as 117 km, and the 3-sigma dispersion in inclination may be readily inferred as 1.92 degrees for a nominal 66-hour trajectory with a radius of closest approach of approximately 120 km.

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[1] JPL TM 312-177, "Guidance Considerations for the Establishment of Lunar Orbits," C. Sauer, May 22, 1962

The principal effect of these uncertainties is to introduce a large variability into the Δv required to circularize the orbit at 250 km. This can range from 1.24 km/sec to 0.97 km/sec over the expected variation in R.C.A., ignoring gravity effects.

2. RETROMOTOR AND AIM-POINT ALTITUDE

A solid-propellant retromotor of fixed impulse is desired for reliability. This retromotor must be sized to cover the maximum requirement (i.e., 1.25 km/sec) including gravity losses, and any excess Δv in a particular situation beyond the requirement for circularization must be dissipated in turning the velocity vector out of the original plane, thus effecting the yaw maneuver discussed by Sauer.[2]

Tradeoffs are possible, wherein orbit altitude is adjusted or widened into a band of acceptable orbits, with the result that the retromotor can be reduced in size. One such tradeoff is illustrated in Figure A, which shows the effect of a variation in aim point, altitude, or Δv with two definitions of design orbit. The recommended retromotor size (1.25 km/sec) permits circularization at 250 km with a probability of 99.74% for an aim-point altitude of 133 km. Up to 3-sigma variations in R.C.A. will be circularized at 250 km with this design.

3. DESIGN ORBIT

The major factor in setting the design orbit altitude is the dispersion in radius of closest approach. Sauer[1] has recommended that the orbit be at least 6 sigma above the surface, or at 234 km. The design orbit altitude of 250 km has been chosen on this basis and satisfies the camera requirements for coverage very well -- as would any orbit up to perhaps 400 km. Lenses for lower orbits are, in general, somewhat shorter in focal length and are therefore somewhat lighter. Otherwise, the orbit altitude choice is not critical until the system parameters are selected.

The yaw maneuver is a necessary consequence of the fixed retromotor Δv and can vary with R.C.A. from 0 to 23 degrees yaw angle for this retromotor and air point. The net effect of this is to vary the inclination and longitude of the nodes of the final orbit.

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[2] JPL Interoffice Memorandum, "Some Results of Using a Fixed Impulse Retro-Rocket for a Lunar Orbiter," C. Sauer to P. Echman, April 27, 1962

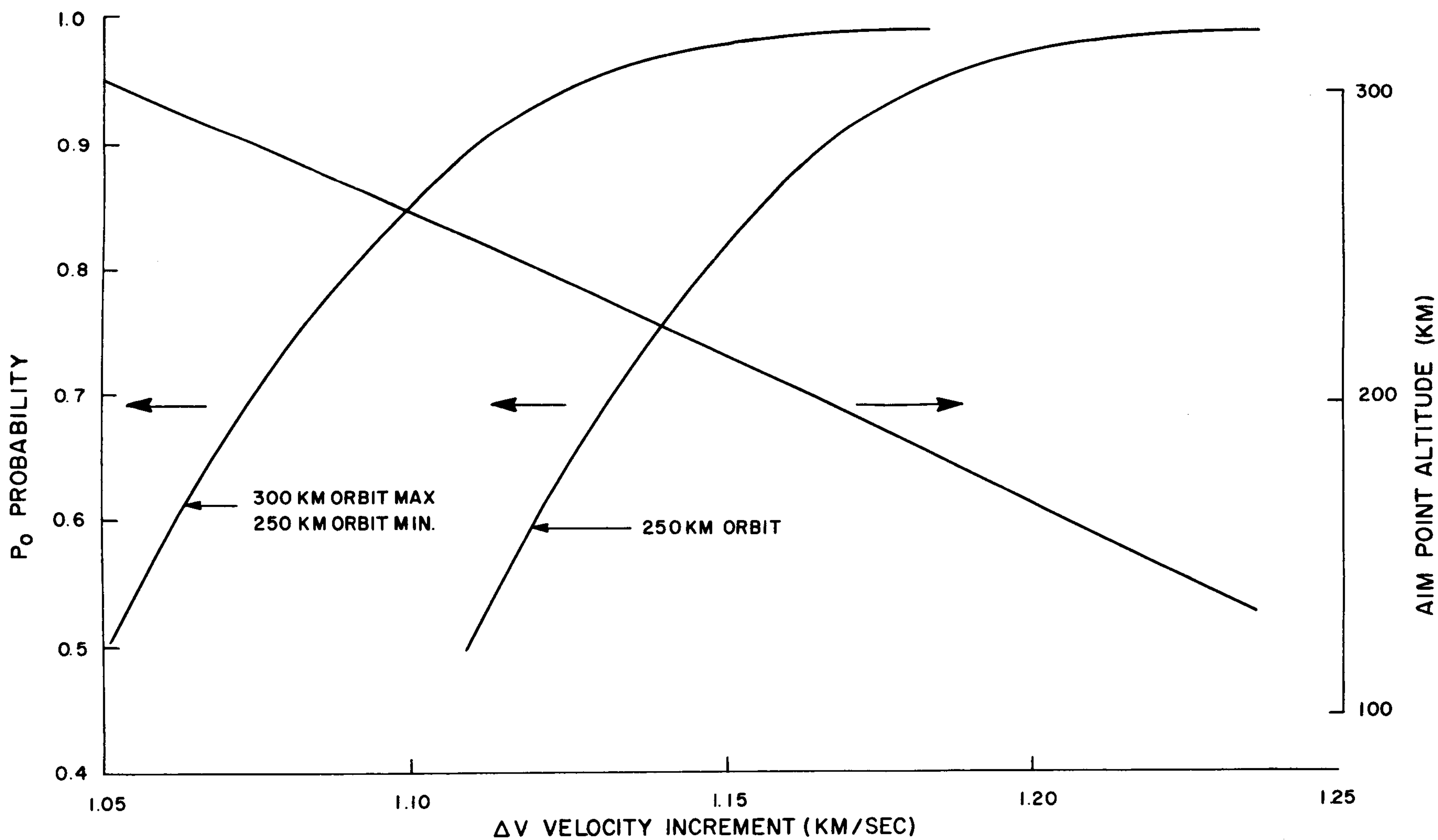

Figure A. Probability and Aim Point Altitude vs. Velocity Increment

There are four maneuvers which provide entry to a possibly satisfactory lunar orbit. Injection may be made from an approach over either the leading or trailing edge of the moon, and the yaw maneuver may be made, in either case, to the right or to the left. Two maneuvers tend to place the nodes near the earth-moon line and can be dropped from further consideration for a first-quarter injection. These are the leading-edge injection, left yaw, and the trailing-edge injection, left yaw.

The trailing-edge injection, right yaw, (toward the equator, since a northern-hemisphere approach is implied throughout) and the leading-edge injection, right yaw, (away from the equator) both tend to place the line of nodes perpendicular to the earth-moon line. The descending node is therefore somewhere near 90 degrees East lunar longitude, depending upon R. C. A., in either case.

The effect of variation in R. C. A. may be seen in Table A. Ascending node longitude and inclination are tabulated against R. C. A. for both maneuvers of interest. An analysis of Table A data suffices to show that the leading-edge injection, right yaw, produces a range of node longitudes 32.1 degrees wide, and the trailing-edge injection, right yaw, a range of node longitudes 18.8 degrees wide. The range of inclinations produced is 12.3 degrees and 16.6 degrees. The trailing-edge injection might be judged the more desirable of the two maneuvers because of the more accurate placement of the nodes of the orbit in the presence of variable R. C. A. The variation of inclination and node with error in incoming inclination is trivial, since the 3-sigma error is 1.92 degrees and the sensitivity coefficients are all close to 1, or smaller.

One additional factor enters into the choice between leading-edge and trailing-edge injections. This is the direction of regression of the nodes, opposite to the direction of regression of the nodes, opposite to the direction of the motion of the satellite. A direct orbit has an inertial rotation of the line of nodes about the lunar axis of approximately 1 degree per day. This is opposite to the sun's apparent annual motion, giving rise to a variation in sun angle near the equator of approximately 2 degrees per day. The leading-edge injection produces a relative motion rate and sun-angle change rate of nearly zero.

If the mission can extend beyond one mapping of the visible face, consideration must be given to the selection of a fixed-sun-angle profile or a time-variable profile. RCA has tentatively chosen to specify the variable sun angle, and therefore the trailing-edge right yaw injection, on the basis that the mission may very possibly include two mappings of the visible face, but not more than this number, so that two different sun angles can be planned for the two mapping intervals. If one should be unsatisfactory, there is a good possibility of obtaining the desired contrast on the other pass. If both are satisfactory, valuable relative height information may be obtained.

TABLE A. COMPARISON OF INJECTION MANEUVERS

A. Trailing Edge Right Yaw			Initial Inclination 27.2° Final Orbit Altitude 250 km	
Radius of Closest Approach	Miss in σ - units	Yaw Angle	Asc. Node E. Longitude	Final Indication
1754 km	-3.0	5.65°	302.98°	28.13°
1812.5	-1.5	12.71	290.39	31.50
1871	0	16.92	285.52	35.02
1929.5	+1.5	20.15	284.15	38.85
1988	+3.0	22.83	289.24	44.71

B. Leading Edge Right Yaw			Initial Inclination 35.55°	
Radius of Closest Approach	Miss in σ - units	Yaw Angle	Asc. Node E. Longitude	Final Indication
1754 km	-3.0	5.65°	305.28°	35.54°
1812.5	-1.5	12.71	293.07	35.63
1871	0	16.92	285.50	35.00
1929.5	+1.5	20.15	279.18	33.35
1988	+3.0	22.83	273.17	27.29

Further work is in progress to define the exact photo coverage available and to devise a coverage sequence for major surface features of interest. A detailed study of photo coverage restraints to the nominal 8-day launch window is also in preparation.

4. PERFORMANCE OF THE LOC TV SURVEYING SYSTEM

The performance of the TV surveying system is determined primarily by the desired resolution and coverage, sensor sensitivity, allowable image motion, communication bandwidth, and achievable orbit and, to a lesser extent, by the power profile of the cameras, the characteristics of the tape recorder, and the uncertainties in the orbit and launch date.

The optimum system performance is achieved by an elaborate tradeoff among the aforementioned factors. The process will be demonstrated and recommendations developed for the TV cameras, optics, and tape recorder parameters.

The primary area of interest lies between 20 degrees north longitude and 20 degrees south longitude on the visible face of the moon; that is, between 270 degrees east lunar longitude and 90 degrees east longitude. Within this area of interest there are various areas such as the vicinity of the craters Kepler and Copernicus and the Ranger 3 through 9 impact areas in Oceanus Procellarum which are of special interest. Since the primary objective of the LOC is to compare various sites with respect to their suitability as Apollo landing sites, it is not necessary to map the entire area between ±20 degrees latitude but only to be able to obtain data within that area for selected sites with a sufficient resolution and coverage so that comparisons of the surveyed sites can be made. In general, the approach to the television survey of these areas leads to two possible choices. Because of limitations in the television subsystem, communications bandwidth, contact time, and power supply, surveying of large areas at the maximum resolution obtainable by the LOC will not permit continuous coverage but only sample coverage of these areas. If the desire for continuous coverage is maintained, then the maximum resolution will be dictated by the requirements of nominal overlap between successive frames along each orbit and the capability of taking frames on adjacent orbits which also overlap. These two conditions are called endlap and sidelap, respectively.

The longitude span to be surveyed and the requirement for reasonably high resolution have dictated the presence of a surface-stabilized camera system. Resolution and coverage, RF bandwidth, and transmission time combine to dictate that the orbit altitude lie within a few hundred kilometers of the lunar surface and be nearly circular. The difficulty in controlling the arrival conditions of the transit trajectory with a single mid-course maneuver and the desire for a minimum-weight retromotor lead to a choice of a minimum orbit altitude no lower than 250 km. Image motion during the exposure for a non-compensating system becomes more severe at low altitudes and/or high resolutions, but is not too severe for most of the cases of interest to the LOC. The 250-km altitude orbit has been found to be a satisfactory design point and is recommended.

With the altitude fixed, the orbit inclination is the dominant factor in setting frame width if sidelap is desired, since the perpendicular spacing between orbits varies with the cosine of the inclination at the equator. Also, the mappable arc along the orbit within the 20 degrees north and south latitude zone is determined by the inclination, although illumination may be a more restrictive constraint to potential coverage extent.

The allowable power, bandwidth, and transmission time fix the number of bits (and therefore the characteristics of the TV scan pattern) and, in combination with the foregoing, the resolution per TV line in the picture. Camera usage when more than one camera is present can admit variation, such as two cameras viewing side by side across the orbital track, or a high and low resolution camera centered on the orbital track. Readout and erase of the camera must be accomplished within the desired frame repetition time, and in general the readout time of the camera will be adjusted so that the readout bandwidth is matched to the bandwidth capability of the tape recorder or video transmitter. Since simplicity is a design goal, a single-speed tape recorder and a single frame rate and readout bandwidth are considered desirable. Existing satellite tape recorder hardware has at least the capability of 62.5 kc signal bandwidth with a 120 kc bandwidth achievable with a moderate development program.

A 1-inch vidicon was chosen as the sensor because of its relatively high resolution, adequate sensitivity, and experience on related programs. The effective number of resolution lines for this camera is approximately 700.

Table B summarizes the investigation of alternative arrangements of cameras, tape recorder, and optical subsystem elements. In general, three possible arrangements have been considered for two possible tape-recorder bandwidths. As is shown the difference in the tape bandwidth does not affect the operational principles of the three basic camera configurations but does vary the total information capability on a per-orbit basis which is commensurate with the increased bandwidth. Of the six configurations, however, only configuration 2 would allow for continuous tape motion all other would operate in a start-stop manner between frame groups.

The present design goals include two vidicon camera systems, which must remain as preliminary until a final tabulation of weights and power requirements for the en tire LOC is determined. Thus, Table B considers the possibility of a single camer system (configurations 1 and 4) which includes the capability of continuous coverag It should also be pointed out that, with a single vidicon and a double set of optics wi some rotating elements, configurations 3 and 6 could also be obtained. This would

TABLE B. ALTERNATIVE TV SYSTEM PARAMETERS

Parameter	Tape Recorder Bandwidth = 60 kc/s				Tape Recorder Bandwidth = 120 kc/s			
	(1) One Camera	(2) Two Cameras Across Orbit	(3) Two Cameras		(4) One Camera	(5) Two Cameras Across Orbit	(6) Two Cameras	
			High Res.	Low Res.			High Res.	Low Res.
Orbital Altitude (km)	250	250	250	250	250	250	250	250
Orbital Inclination (degrees)	35^o	35^o	35^o	35^o	35^o	35^o	35^o	35^o
Field of View (km)	30	16	6.5	54	30	16	6.5	54
Number of TV Lines	650	650	650	650	650	650	650	650
Resolution/TV Line (meters)	45	25	10	85	45	25	10	47
Camera Read Out Time (seconds)	4.78	4.78	4.78	4.78	2.48	2.48	2.48	2.48
Tape Start-Stop Time (seconds)	1.00	0	1.00	1.00	1.00	1.00	1.00	1.00
Total Record Time/ Frame Group (secs)	5.78	9.56	10.56	10.56	3.48	5.96	5.96	5.96
Number of Frames (Total)	83	100	90	90	137	160	160	160
Separation Between Frames centers along orbit (km)	21.1	13.3	49	49	31.1	13.3	27.8	27.8
Linear Coverage (km)	1750	665	2220	2220	2900	1060	2220	2220
% Coverage Between ± 20^o Lat.	79	30	100	100	132	48	100	100
Time Between Frames (seconds)	15.4	9.58	35.8	35.8	15.4	9.58	20	20
Erase Time (seconds)	10.6	4.78	>20	>20	12.9	6.8	>15	>15
Total Survey Time (mins)	21.2	8.1	27	27	27	13	27	27
Allowable Exposure Time for 1 TV Line Smear and 1.0 millirad/sec Attitude Stabilization (ms)	27	15.2	6	57	27	15.2	6	28.6
Effective Focal Length (mm)	93	175	430	50	93	175	430	93

lead to some differences in operation, but the coverage and resolution as shown for configurations 3 and 6 could be obtained with a single vidicon.

Configuration 2 is recommended as the best choice at this time in terms of reliability, resolution, coverage, constraints of schedule, power and weight, and development cost. A difference of only a factor of 2 to 3 in resolution over configuration 2 is obtained by going to the highest resolution of these configurations (10 meters) and should not lead to a tremendous improvement in the amount of surface characteristic data extractable from the television frames. Furthermore, the fact that contiguous coverage at the high resolution would not be obtainable with configurations 3 and 6 might hinder the interpretation of the data, since both coverage and resolution are important parameters in the data reduction process. In case two vidicons are not within the capability of the early LOC, reconsideration of this choice may be dictated.

A catadioptric lens system of f/1.5 has been selected and, for a camera sensitivity of 0.2 ft-candle-seconds at a S/N of 32.5 db, this will permit observations of the lunar surface with a minimum albedo of 0.09 and one TV line of smear per exposure due to attitude motion rates of 4×10^{-3} rad/second at sun angles of approximately 60 degrees (i.e., 30 degrees from sun terminator). This lens will permit observation of albedos to 0.06 with a 29-db S/N ratio and sensitivity of 0.1 ft-candle-second at the same sun angles (see Table C).

A summary of the parameters of the TV system, and expected performance are:

Orbit:	Altitude	250 km
	Inclination	35 deg
	Surface Speed	1.37 km/sec
Camera:	Video Bandwidth	62.5 kc
	Number of Cameras	2
	Field of View per Camera	18 km square
	Combined	18 x 34 km
	Scan Lines	1,024
	Resolution Lines	700 vertical
		695 horizontal
	Readout Time	6.0 sec
	Erase Time	6.0 sec
	Frame Repetition Rate	12.0 sec
	Mapping Time	8 min. or 480 sec
	Mapped Arc	660 km
Lens:	Focal Length	153 mm
	f/n (max.)	1.5
	Aperture	100 mm
	Exposure Time	17 msec
	Smear	1 TV line/exposure
	Smear Rate	4.0 milli rad/sec

TABLE C. f-NUMBER vs ATTITUDE STABILIZATION RATE

Attitude Stabilization Rate	Exposure* Time (millisecs)	Sun Angle Degrees	Photometric Function	f-Number for: S/N = 32.5 db Exp. = 0.2 ft-cand-secs.			f-Number for: S/N = 29 db Exp. = 0.1 ft-cand-sec.
				albedo = 0.18	0.09	0.06	albedo = 0.06
1.0×10^{-3} rad/sec	15	60	0.22	2.4	1.7	1.4	2.0
2.0×10^{-3} rad/sec	13	60	0.22	2.3	1.6	1.35	1.9
3.0×10^{-3} rad/sec	12	60	0.22	2.2	1.5	1.25	1.75
4.0×10^{-3} rad/sec	10	60	0.22	2.0	1.4	1.10	1.60

* Exposure time based on 1 TV line smear including motion of vehicle over the lunar surface of 1.37 km/sec and a lunar orbit altitude of 250 kilometers.

Tape	Tape Speed	30 inches/sec
	Motion	Continuous
	Length	1200 feet
	Number of Frames	80 total
	Total Recording Time	8 minutes
	Total Playback Time	8 minutes

C. RECOMMENDED SYSTEM CONFIGURATION

1. RETRO-PROPULSION SUBSYSTEM

A solid-propellant rocket with spin stabilization during retro-firing is recommended for the LOC. The proposed motor is a 3000-pound-thrust unit with a vacuum specific impulse of 280 seconds and a total velocity increment of 1250 m/sec. Spin-up would be accomplished with a system similar to the Ra 3, 4, 5 solid-propellant spin system. The spin rate during retro-firing will be approximately 300 rpm.

2. ATTITUDE CONTROL SUBSYSTEM

A conventional three-axis jet-controlled system operating in a limit cycle mode which maintains the LOC principal axes coincident with moon-centered axes for at least 1 month in lunar orbit is recommended. Roll and pitch errors are detected by a two-axis horizon sensor, and yaw error by either a two-axis gyro-compass or by a roll/yaw rate-gyro combination. A torquing subsystem of six thrust nozzles, two per axis, with a common regulator valve at a 0.01-pound thrust is proposed. An operational lifetime of greater than 1 month should be achievable with such a subsystem. This subsystem is selected despite its disadvantages of high power requirements and difficult development schedules primarily because it offers a greater degree of flexibility and growth potential. Furthermore, such techniques as developed for this subsystem would be directly applicable to later lunar orbiters, including manned systems. An over-all accuracy of stabilization of ±1 degree with stabilization rates no greater than 0.01 deg/sec is recommended.

3. TELECOMMUNICATIONS SUBSYSTEM

The telecommunications subsystem recommended would consist of three major units, all operating at L-band frequencies. These units would comprise a tracking transponder (two-way, phase-locked doppler system) similar to the system presently used on the Ranger spacecraft, a 20-watt video transmitter similar to the video communications system presently on the Ra 6-9 TV capsule, and a command subsystem compatible with the DSIF command capabilities and utilizing as much Ranger command circuitry as practicable for the LOC. An omnidirectional antenna providing look angles of the order of 100 degrees in elevation and 360 degrees in azimuth will be utilized with the transponder and command units. A directional antenna of approximately 30 degrees beamwidth and a 15-db gain on axis will be used for the video transmission link. The directional antenna will be trainable in elevation on ground command approximately ±35 degrees about its nominal orientation in suitable steps to assure at least 10 minutes/orbit contact time with earth stations. A video bandwidth of no less than 62.5 kc is recommended. Telemetry information within the capability of the transponder and video transmitters to accommodate such information is proposed. In general, a low-power telemetry subsystem employing only two subcarriers on the transponder is recommended, with priority being given to that telemetry associated with the operating status of the capsule and necessary for predicting the future capabilities of the LOC for operational programming.

4. TELEVISION SUBSYSTEM

The recommended television subsystem consists of two 1-inch electromagnetically focused and deflected vidicons and the associated camera circuitry. A resolution of approximately 25 meters per TV line with a total of 700 resolution lines per TV camera is proposed. The necessary optics with an effective f-number of approximately 1.5 will be utilized so that mapping of the lunar surface at sun angles between 30 and 60 degrees with a reasonable sensor sensitivity is obtained. Iris control by ground command is also proposed to provide for optimum utilization of the camera subsystem under the wide dynamic range of surface illumination characteristics likely to be encountered. The camera subsystem should be capable of providing both direct and remote operations on ground command. In order to obtain remote operation, a video tape recorder is required to record the video information obtained while the LOC is not in a position for transmission of the video data to the earth. In general, the area where video transmission may take place will occur between ±15 degrees of lunar longitude centered about the nominal earth-moon line. Tape recorder bandwidth will be at least 62.5 kc, with a capability for continuous recording of the video data for approximately 8 minutes, or 100 TV frames of approximately 1000 scan lines per frame, having a horizontal resolution of approximately 700 TV lines.

5. POWER SUPPLY

The power supply subsystem must be capable of supplying the necessary constant loads required by the attitude control subsystem and the peak loads required by the TV and telecommunications subsystems. In general, the constraints on the LOC would prohibit the use of a solar-cell array only without foldable panels.

A power system utilizing either a combined radioisotope thermoelectric generator (RTG) and solar-cell array or an all-RTG supply is recommended. During the TV operational phase of the LOC, an average output of 75 watts unregulated on a continuous basis is recommended, with a capability of handling peak requirements of approximately 190 watts for short periods.

SECTION II

RETROPROPULSION

A. GENERAL CHARACTERISTICS

The subsystem for injection of the capsule into an orbit about the moon from the transit trajectory consists of a solid-propellant rocket with spin stabilization during firing. Spin-up is accomplished with the Ranger 3, 4 and 5 solid propellant spin system.

B. SOLID PROPELLANT MOTOR

The recommended design is a 3000-pound thrust solid-propellant rocket with the following design characteristics:

(1) Spherical casing

(2) Propellant vacuum specific impulse of 280 seconds and density of 0.63 lbs./in^3, arranged in 8-point star grain configuration

(3) Conical semi-recessed graphite silicon-phenolic nozzle

(4) Asbestos Buna-N insulation liner

These recommended design characteristics were based upon the following pertinent points:

a. A thrust level of 3000 pounds was selected because:

   (1) The motor size is compatible with the capsule dimensional constraints.

   (2) Steady state accelerations are approximately the same as experienced during earth launch.

   (3) The resultant burning time is compatible with design technology and operational condition.

b. The propellant specific impulse and density necessitate no development program.

c. A spherical shape was selected over a cylindrical shape because:

   (1) The motor is 3 inches shorter which is in the right direction when considering boost vehicle moments and capsule structural weight.

(2) Lower motor inert weight results because of an improved structural factor (0.108 vs. 0.13), which is in the right direction with payload weight as a critical item.

(3) The spherical design is a developed item.

d. In consideration of solid rocket design constraints, modification of the Ranger 3, 4 and 5 retromotor would result in cost and schedule very close to that of a new design for the following reasons:

(1) The greater than 20 pounds decrease in propellant weight would require a grain configuration development effort.

(2) Different attachment would require a case development effort.

(3) Changes (1) and (2) are of sufficient scope to necessitate a full flight proof test program.

TABLE 1. COMPARISON OF ROCKET MOTORS FOR CAPSULES HAVING A GROSS WEIGHT OF 400 AND 450 POUNDS

Pertinent Factor		400 Pounds	450 Pounds
Payload Weight	(pounds)	232.9	261.9
Motor Weight	(pounds)		
Total		167.1	188.1
Propellant		149	167.8
Inert		18.1	20.3
Motor Shape		sphere	sphere
Diameter Motor	(inches)	17.36	18.26
Total Length Motor	(inches)	27.86	28.76
Burning Time	(seconds)	13.63	15.37
Total Impulse	(pound-seconds)	40,900	46,100
Acceleration	(g's)		
Initial		7.5	6.67
Burnout		11.8	10.5
Ignition		less than 40	less than 40
Rate of on-set	(g's/second)	150,000	150,000

The comparison of the solid propellants was based on the following fixed conditions:

Velocity Change	1250 meters/second
Thrust	3000 lbs ±10%
Vacuum Specific Impulse	280 $\frac{\text{pounds-seconds}}{\text{pounds}}$
Thrust Coefficient	1.867
Chamber Pressure	600 PSiA
Nozzle Expansion Ratio	24
Structural Factor (includes mounting and ignitor)	0.108
Nozzle Half Angle	16 degrees
Sliver Loss	2%
Total Impulse Uncertainty	1%
Volumetric Efficiency	92%

C. MAJOR INTERFACES

1. LOADS

The rocket motor case will be used to transmit the dynamic and static loads from the capsule to the Ranger Bus.

2. ARM

The ignition system must be armed prior to capsule separation. Ranger safety rules will necessitate that this event occur after "Agena" first burn to eliminate the requirement for a destruct subsystem.

3. FIRING TIME

The point in time at which the motor is fired must be varied to compensate for the change in motor performance as a function of propellant temperature. This will be accomplished prior to separation of the capsule from the Ranger Bus.

4. ORIENTATION

The capsule must be properly oriented prior to firing of the retromotor.

5. CAPSULE SEPARATION

The capsule must be separated from the Ranger Bus prior to retro firing.

6. SPIN-UP

The capsule must be spinning during retro firing to maintain thrust misalignment requirements within acceptable limits.

7. FIRE

Initiation of firing will necessitate a 28-volt signal.

8. MOTOR SEPARATION

The motor will be separated from the capsule by the application of a 28-volt signal to the separation device. The separation velocity must be sufficient to prevent collision between the capsule and burnt motor case and will vary according to the installation configuration. Also, separation must be delayed sufficiently to prevent collision as a result of motor outgassing after burnout. Delay time will be from 3 to 6 minutes. The maximum motor external temperature will be 400° F at the nozzle and will be developed approximately 1 minute after burnout.

9. EXHAUST PLUME

The plume probably will not cause excessive capsule temperature but may cause solar cell performance degradation because of metallic deposit. This item will necessitate altitude test at AEDC.

10. TEMPERATURE LIMITS

The motor temperature must be maintained between 40° F to 90° F.

D. STATUS AND FUTURE PLANS

The solid rocket motor performance and physical characteristics developed up to this time were evaluated by the Elkton Division of the Thiokol Chemical Corporation. Upon configuration freeze, the design will be submitted for review to the Thiokol Chemical Corporation, Hercules Powder Company and Rocketdyne.

E. SPIN ROCKETS

Preliminary analysis indicates that the Ranger 3, 4 and 5 spin-rocket installation would be sufficient. Thus, this system, which consists of a simple gas generator (0.6 pounds propellant) supplying three symetrically disposed torque nozzles is recommended. In view of this recommendation, a very low effort was expended in this area during this configuration freeze period.

SECTION III

ATTITUDE CONTROL SUBSYSTEM

A. INTRODUCTION

The effort during the first month of this study has been restricted to those areas which most critically effect the selection of the system to be evaluated in detail during the latter part of this phase of the study, that is, those areas which lead to a firm comparison between the merits of Systems A and B (the pure gas and pinned system, respectively). The study has not been restricted solely to the technical features of each system and their growth potential, but covers also the more practical aspects of development time and cost, state of the art, and direct RCA experience in the field. A large part of the effort has been taken up in discussions with possible subcontractors and in assessments of their proposals. Another major task has been to evolve the configuration for System B in sufficient detail so that accurate weight (particularly structural weight) and power estimates could be made. It was necessary also to establish the feasibility of satisfying the fairly stringent conditions on the moments of inertia*. However, the configuration and mechanical design problems, being more closely related to the structural subsystem, are discussed later in this section.

Before discussing the problem areas in each system that have been studied since completion of the first phase, for completeness, a brief description of each of the systems is given below. With minor modifications these descriptions have been taken from the Phase I report.

B. DESCRIPTION OF SYSTEMS

1. SYSTEM A — PURE JET SYSTEM

System A is a conventional 3-axis jet-controlled system operating in limit cycle mode (which maintains the satellite principal axes coincident with moon-centered axes.) Roll and pitch errors are detected by a 2-axis horizon sensor, and yaw errors by either a 2-axis gyrocompass or by a roll/yaw rate-gyro combination. The torquing subsystem consists of six 0. 01 lb. -thrust nozzles, 2 per axis, with a common regulator valve. On the basis of total impulse requirements for one month's operation (computed both before and during the first month of this study), a cold gas (nitrogen) system with an impulse of 228 lbs. -secs. (4 lbs. of nitrogen) has been tentatively selected. The total weight of this system is 14 lbs. , including regulator, nozzles etc.

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* Both Systems A and B have been analyzed in greater detail than was System A in the Phase I report.

Although a particular technical feature of the gyrocompass, namely that it provides yaw error explicitly, would indicate a preference for this sensor, the estimates of development time for such a gyro that have been obtained from manufacturers were not considered to be compatible with the development schedule for LOC. For this reason effort in the computer study is now being redirected to determine the characteristics of the horizon scanner rate-gyro system. The minimal system consists of one two-axis scanner and two rate-gyros (one roll and one yaw). This presupposes that derived-rate feedback in the pitch channel is satisfactory, which, for the limited total impulse available, has not yet been established. At worst, three rate-gyros will be required.

The study has clearly shown that the horizon sensor presents a more critical design problem than does any other stabilization equipment item. However, as a result of extensive discussions with potential manufacturers held since publication of the Phase I report, RCA is able to submit better estimates of weight and power requirements and development schedules for these units.

2. SYSTEM B — PINNED SYSTEM

In order to overcome (or at least reduce considerably) problems presented by limited knowledge of the performance of horizon sensors, an alternative three-axis control system was evaluated. This system employs two passive (static) horizon detectors, unlike the rotary or nutating devices generally required for the convential system, and requires no yaw sensor at all. Passive units require development also, but the predicted weights and, more important, the predicted power requirements for these units are substantially lower than those for other types.

This second method of stabilization, which could be described as a modified spin-stabilization system, is sometimes termed "pinned system" because, like a spinning rigid vehicle, one body-axis is "pinned" or fixed by the inherent gyroscopic rigidity of the vehicle. The difference between this system and a passive spin-stabilization system is that, in this system, the high angular momentum necessary to achieve gyroscopic stability is stored in a "flywheel", the axis of which is parallel to the satellite axis of symmetry. Thus, the main body itself can be non-rotating and yet retain the stability associated with spinning bodies. The flywheel may contain all equipment which does not form a part of, or which does not have to be rigidly attached to, those subsystems requiring orientation, such as for this system, the camera and antenna subsystems. In fact, the larger the rotating part, the lower the angular velocity need be for a given stiffness. Moreover, the lower the relative angular velocity, the longer the bearing life and the greater the economy in power.

In System B, the solar cells, batteries, spin-axis precession jet system, and the horizon sensors constitute the flywheel. The principle of the pinned system should be clear from Figure 1. This diagram is schematic in form and is not to scale.

The horizon sensors are attached to the rotating component, and their optical axes generate two cones with a common axis coincident with the satellite spin-axis. The intercepts of these cones with the lunar surface are shown in Figure 1. If the spin-axis is normal to the orbit plane (the desired orientation), the lengths of these intercepts

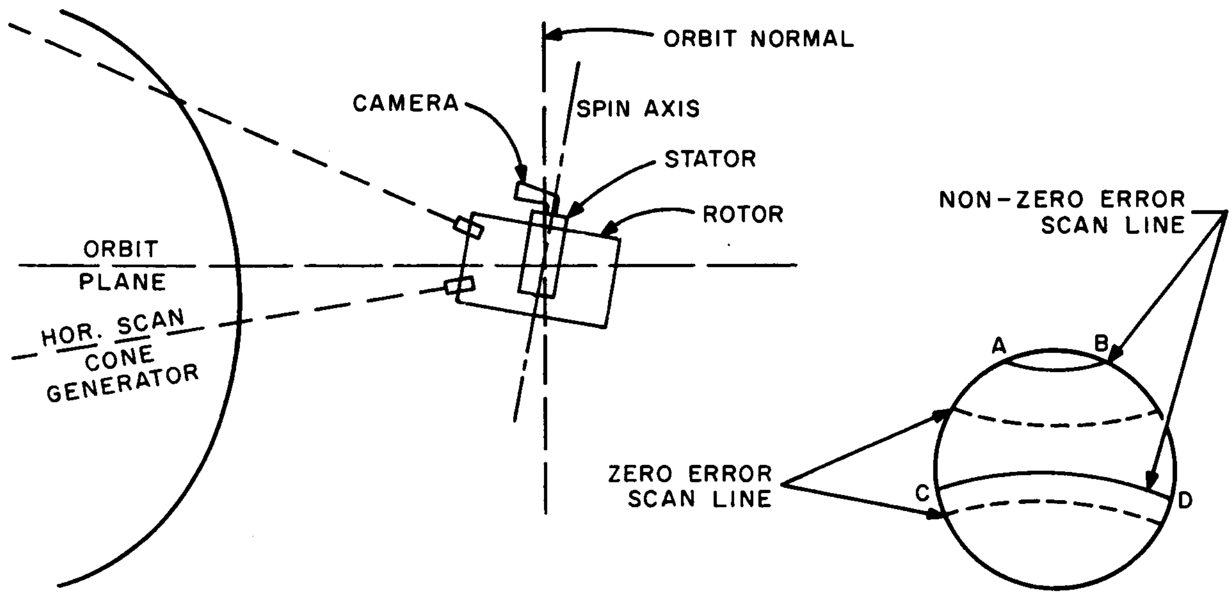


Figure 1. Spin Axis Error Sensing

(or the "moon times") are equal. However, should the vehicle develop a roll error, as shown in Figure 1, the lines will be of different lengths (AB and CD). Should closed-loop control be desired, a relatively simple comparator circuit can be devised providing a d-c output proportional to roll error. Yaw error cannot be determined from horizon sensor output, but knowledge of instantaneous yaw error is not required in this system. A pure yaw displacement (in the rotating reference frame) will appear as a pure roll displacement one-quarter of an orbit later. Therefore, continuous or intermittent correction of the roll displacement will be sufficient to maintain the desired orientation of the spin-axis. The drift rates resulting from natural disturbing torques will be very small and certainly less than the apparent drift caused by the regression of the orbit plane; the combined rates do not exceed 2° per day. Thus, considerable time delays are tolerable between error detection and application of control torque. Consequently, it will be possible (and probably preferable) to employ open-loop control of the spin-axis attitude. An alternative method by which spin-axis attitude error (and not simply its roll component) can be determined rapidly, utilizes sun-aspect information obtained from a solar digital aspect sensor in conjunction with the horizon detector outputs. This technique, presently being developed for the NASA/RCA Relay satellite, is recommended as a means of determining attitude subsequent to spin-up at the end of the three-axis control mapping phase of the mission. This technique can also be used in the pinned stabilization system if simultaneous correction of roll plus yaw errors (total error) is desired.

Because impulsive torques applied to a spinning body give rise to displacements and not, as for stationary bodies, to angular velocities, a very simple torquing system can be employed which delivers impulses of fixed magnitude. This technique, the so-called double-pulse system, is described in some detail in the Phase I report. By correctly time-sequencing the firing of a pair of pulse-jets it is possible to rotate the axis of a spinning body without inducing residual precession.

Control about the spin axis can be achieved by modulating the flywheel speed as in a conventional three-axis flywheel system. Unless the total satellite angular momentum is subject to large variations, no pitch-jet control will be required. Unlike the spin-axis attitude control subsystem, the pitch control channel would have to be fully automatic. A signal obtained by splitting the horizon (that is, by bisecting either lines AB or CD in Figure 1) would be compared with one derived from a reference on the stationary component. In this way, it is possible to generate a d-c voltage proportional to pitch error so that a standard proportional control loop can be employed.

C PROBLEM AREAS

1. SYSTEM A

As stated earlier the horizon scanner presents the most critical problem in the pure jet system.

Information received from the major commercial producers of horizon sensors indicates that there are no tested or developed sensors for Lunar missions available. RCA has approached the following companies on this matter:

Advanced Technology Laboratories	Mountain View, Calif.
Avion	Paramus, N. J.
Baird-Atomic	Cambridge, Mass.
Barnes Engineering	Stamford, Conn.
Eastman Kodak	Rochester, N. Y.
Kollsman Instruments	Elmhurst, N. Y.
Minneapolis-Honeywell	Los Angeles, Calif.
Perkin-Elmer	Norwalk, Conn.
RCA	Burlington, Mass.
Servo Corporation	Hicksville, N. Y.
H. R. B. Singer	State College, Pa.
Texas Instruments	Dallas, Texas

The most promising proposals to date have been forwarded by Advanced Technology Laboratories and Barnes Engineering. The former proposes a modified version of the nutating-type horizon tracker under development for the Gemini project. The latter manufacturer proposes a conical-scan technique that requires 2 scan-heads for 2-axis information. Barring any new development it is probable that one of these two sensors will be recommended for the pure jet system. However, being mechanical scanners, each requires a continuous power input in the range from 8 to 15 watts which is a heavy demand on the power supply of a vehicle the size and weight of the LOC. For this reason, the possibility of devising a passive sensor for this application is being examined. One scheme which incorporates a detector mosaic shows some promise if resolution requirements can be relaxed. However, due to the low minimum temperature (123°K) of the moon's surface a sensor of this type would require prohibitively large optics to achieve the same resolution as a scanner. Another scheme, incorporating a torsion-spring mounted mirror and a passive detector is also being considered. A further problem associated with the horizon sensor is one of development times. Barnes Engineering estimates 12 to 15 months for delivery of the first Flight item. Although A.T.L. has not yet given a firm estimate, its schedule is likely to be the same. Therefore the earliest flight date, allowing three months for systems integration and test, might well be 18 months from award of subcontract.

The other serious problem area in System A is one of total power requirement. The current estimates (allowing 12 watts for the horizon sensor) is 22 watts including a 2-watt contingency on the scanner. No power is allowed for gyro heaters, but, as high linearity is not required, their temperature can be controlled to sufficient accuracy by the satellite thermal control system. As stated earlier a third gyro may be required if derived pitch-rate proves too noisy. Thus the total power requirement could increase to 25 watts.

Excluding the scanner no serious development problems are anticipated. From results obtained so far from the computer study, the total impulse allowed for the system appears to be adequate. The error rates achievable with this system, being less than 0.01°/sec., are of course in no way critical with respect to camera "smear" requirements. For a mechanical scanner, pointing accuracies of approximately 1° are anticipated.

2. SYSTEM B

The problems on System B have arisen mainly in the area of the structural and mechanical system. These are considered elsewhere in this report. There are no serious problem areas in the attitude control system itself provided the residual error rate estimate given below is considered acceptable. The relative importance of this particular parameter determines the merit of System B.

a. Horizon Sensors

The horizon sensor problem is greatly simplified in this system as the unit itself requires no moving parts. Preliminary calculations carried out by Barnes Engineering have indicated that it would be feasible to modify one of their existing horizon pulse generators. They have estimated 4-1/2 months to delivery of an unqualified unit. Qualification testing would be carried out by RCA; this is exactly the same procedure that has been adopted for other AED projects. Based on this experience RCA estimates not more than two months would be required to complete these tests. Thus the total time to delivery of a flight unit would be less than half that for System A.

b. Inertia Conditions

Due to the dimensional constraints imposed by the Ranger Bus geometry, it has been particularly difficult to realize the conditions on the moments of inertia. The conditions for passive stability of the satellite are:

(1) Damper mounted on the stator:	Stable for all values of the inertias.
(2) Damper mounted on the rotor:	Polar moment of inertia of rotor alone greater than total transverse inertia.

It would appear that the problem would be solved by mounting the damper on the stator. However the decay time-constants attainable with a stator damper are far greater than those for a rotor damper. In addition, the RCA passive "Team" damper proposed for the system will not work at all if mounted on the stator. It is now anticipated that the small residual precession angle required to minimize camera smear can be realized using this device (with increased cart mass) rather than a gyro damper, which of course is less reliable and consumes power. Lastly, flexure of the rotor will dissipate energy and if condition (2) is not satisfied this will tend to increase the precession. For these reasons every attempt has been made to satisfy condition (2).

c. Asymmetry

Due to the presence of the tape recorder and rotatable antenna on the stator it will be extremely difficult to maintain the polar principal axis of this component parallel with the bearing axis. Center of gravity shift will also affect the dynamics but, by careful design, this parameter will be far easier to control. An analysis of the former effect, however, has shown that it does not have a serious influence on the satellite dynamics. Principal axis displacements of 20° to 30° have been shown to be tolerable.

d. Dynamic Balancing

Information from various sources has been accumulated on the dynamic unbalance tolerance that can reasonably be specified for this system. The specifications for the balancer that RCA will shortly purchase have been taken as a basis for the assessment. However, experience has shown that satellite flexure gives rise to greater inaccuracies than machine errors. It appears that the minimum detectable angle between nominal and principal axis of inertia will be between 0.20 and 0.25°. It is this error which gives rise to the (relatively) high residual rates associated with System B. Two methods can be employed to overcome this problem. First, the camera itself can be gimballed. This is not too serious a problem as the amplitude (0.25°) is not large. Second, as the rate error due to precession varies sinusoidally at precession frequency, shutter operation could be phased to coincide with the minimum rates. With this latter system, rates of approximately 1 mr/sec. would be achieved. An additional rate-gyro would be required. Both of these techniques will be considered further in the growth potential study. In the absence of compensation a residual rate of approximately 4 mr/sec. is considered to be the lowest achievable with this system. Pointing accuracy would be the same as for System A.

e. Pitch Servo

It is conceivable that realization of the very low error-rates required of the system may complicate the design of the pitch control servo. The performance of this low rate sampled-data system depends on the fluctuation, if any, in the friction torque between rotor and stator. However this problem can be overcome by employing a pitch-rate gyro in conjunction with the horizon-splitter. To account for this possibility, the power estimate given in Table 2 includes a 3-watt allowance for this unit. Only one motor is now considered necessary for this loop.

TABLE 2. POWER REQUIREMENT FOR SYSTEM B

Motor	4 watts
Pitch gyro	3 watts
H. P. G's circuitry	2 watts
Contingency	1 watts
Total	10 watts

D. SUMMARY OF ATTITUDE CONTROL SYSTEM STATUS

On the assumption that the inertia conditions are realized on System B and that stabilization power requirements do not rule out System A, the main basis for comparison is the trade-off between scanner development for System A and residual error rates for System B. The choice to a large extent will depend on the relative merits of these two factors. System B is presently thought to have greater reliability and would hold a slight advantage in a failure mode analysis, as the transition from mapping phase to ranging phase is simpler in this case. However this transition is not considered a very important factor.

Table 3 summarizes the relative merits of each system.

TABLE 3. COMPARISON OF SYSTEM A AND SYSTEM B CONFIGURATIONS

Characteristic	System A	System B
1. Reliability	Major problem is electro-mechanical components involving rotating scanners, logic electronics, and valving for jet operations. Complexity of control systems is considered to be greater than that for System B, (i.e., more valves, electronics and a more complicated horizon sensor).	Major reliability considerations are the operation of a bearing system in a vacuum, the necessity of distributing raw power (DC) and simple control signals through slip-rings and the electro-mechanical elements of the control system. In general, most of the problems are mechanical in nature and the general electronic control circuits are simple. System B also has a better failure mode consideration in that a major part of the attitude control system is passive and failure to achieve local vertical orientation would not cause complete loss of the mission.
2. Development Problems	The primary problem is the development of the horizon scanner. Present data would indicate as much as 15 months to integration of horizon scanners into the first flight units, leading to about 18 months to delivery of the first flight units.	The major problems are: Design of bearings for operation in a vacuum, electrical coupling of raw power (DC) and simple control signals through slip-ring contacts, control of the system configuration for proper inertia distributions between rotor and stator, and maintaining dynamic balance of the system. The estimated development time to complete integration of the attitude control system into a flight unit is 12 months leading to about 15 months for delivery of the first flight unit.
3. Power and Weight Requirements	The estimated regulated power requirements are 22 watts on a continuous basis. The weight of the attitude control elements is approximately 52 pounds.	The regulated power requirements are 10 watts on a continuous basis and the weight of the attitude control system approximately 32 pounds.

TABLE 3. COMPARISON OF SYSTEM A AND SYSTEM B CONFIGURATIONS (Cont)

Characteristic	System A	System B
4. Margin	System A is beyond the margins allowable in weight and power. The continuous power load leads to an inordinately large solar cell array which is constrained by the volume allowable between the Ranger Spacecraft panels. In order to obtain more effective cross-section area fold out panels or sun oriented panels would be required. The weight margin could be achieved by cutting back to one set of vidicons and optics.	Presently the system is marginal in achieving proper inertia distribution and is slightly greater than the allowable weight for the capsule. Both of these problem areas may be allieviated by cutting to a single set of vidicon and optics and adding weight to the rotor portion of the capsule. It now appears certain that, in order to achieve the desired inertia distribution, extending arms will have to be provided.
5. System Flexibility	On a basis of subsystem design and modification as well as growth this system permits a greater degree of flexibility. The primary consideration would be in overall location of the center of gravity which should not present a major design problem.	Changes in subsystem design could only be made on a basis of the effect it might have on inertia distributions and as a result might create a major design problem. This system, however, offers a greater operational flexibility since ground control of the spin-axis orientation is presently included in the design, permitting pointing of the optical axis in at least one plane.
6. Stability Tolerances	Absolute pointing accuracy to the local vertical ± 1 degree. Rotational rates introduced to optical system by attitude control less than 0.01 degrees per second.	Absolute pointing accuracy to the local vertical ± 1 degree. Maximum rotational rates introduced to optical system by attitude control approximately 0.25 degrees per second.

TABLE 3. COMPARISON OF SYSTEM A AND SYSTEM B CONFIGURATIONS (Cont)

Characteristic	System A	System B
7. Growth Potential	Because of the low rates of rotation introduced by the attitude control imate, motion compensation or higher TV sensor sensitivities would be the only limitation on resolution capability, and these techniques could be designed within the system with little change in the attitude control.	In order to take advantage of image motion compensation or higher TV sensor sensitivities, means for stabilizing the camera system would be required over and above the capsule stabilization. This may require that the camera plus optics be mounted on a single axis gimbal or that shutter operation be phased with precession frequency.
8. Lifetime	It is anticipated that lifetimes greater than 30 days can be achieved with a nominal supply of control gas.	Lifetimes much greater than 30 days can be achieved with a nominal supply of propellant for spin-axis control.
9. Cost	In terms of development cost a major expenditure for the horizon scanner would be necessary. A total cost near one million dollars would be encountered in this area.	Because of the lower requirements on the cost of the horizon scanner, development cost for the system would be considerably smaller. Some additional cost would be encountered in bearing test and overall structure and electrical design over and above system A requirements.
10. Related Experience	Most of the system operation would be quite similar to the operations of attitude control systems being utilized on (a) the Ranger Spacecraft, (b) Discoverer satellites, and (c) Agena booststages.	The system is quite similar to the technique employed successfully on the Orbiting Solar Observatory. The TIROS satellites embody certain principles of the technique but with different mechanisms. Control of the spin-axis is obtained by torques generated by the interaction of the earth's magnetic field and an artifically produced magnetic field on the satellite.

E. MECHANICAL AND STRUCTURAL CONFIGURATION

1. INTRODUCTION

Since the completion of the Phase I portion of this study, further detailed investigation has revealed the necessity for increasing the original estimate of power supply requirements for LOC. The specific details of this necessary increase will be discussed in Section VI of this report. Of interest at this particular point are the effects of the increased power requirements on the mechanical and structural configuration of the system.

As a result of the increased power requirements, it was necessary to reconfigure the pure-jet attitude control subsystem (System A). For purposes of this Phase II study, three preliminary configurations will be presented for this system. They may be summarized as follows:

(1) A power supply consisting solely of 15 sq. ft. of projected solar cell area plus batteries.

(2) A power supply consisting solely of a 75-watt radioisotopic thermoelectric generator (RTG) plus batteries.

(3) A combination power supply consisting of a 30-watt RTG and 8.3 sq. ft. of projected solar cell area plus batteries.

The RTG is more adaptable than the solar array as far as the configuration is concerned. On the other hand, provisions must be made to remove the thermal energy during the launch mode. However, because of the low temperature of the RTG, this is not expected to be a problem.

Because of the lower power supply requirement of System B (pinned system), only one power supply consisting of a 8.5 sq. ft. array with batteries will be presented.

A general comparison between systems A and B is presented in Table 4.

2. SYSTEM A - GENERAL DISCUSSION

In general, System A is the simplest and most efficient from a mechanical and structural standpoint. As explained previously, due to the increased requirements, a heavier load must be carried by the structure.

Each of the System A configurations will be detailed and compared with each other in a subsequent section.

TABLE 4. GENERAL MECHANICAL COMPARISON BETWEEN SYSTEMS A AND B

	SYSTEM A	SYSTEM B
Weight	Weight marginal as result of increased power requirements for a non-spinning satellite.	Weight budget provides additional weight to attain desired inertia ratio.
Reliability	Minimization of moving parts - less reliability as result of continual operation of jets - may require redundancy.	Less reliability from the standpoint of requirement of bearings, slip rings and gears and unlocking of the stator and rotor as well as weight erection.
Flexibility	More adaptable to changing design and future missions - fewer constraints.	Additional constraints imposed by the above. Flexibility of design limited considerably.
Design Simplicity	Straightforward mechanical design.	More complex design as result of designing and testing bearings, slip rings, motor drive systems, locking mechanisms and weight erection system.
Principle Axis Shift	None (IF antenna support at the c. g.)	Can accomodate shifts up to 20^{o} to 30^{o}.
Proper Inertia Ratio	Readily achieved inertia ratio defined by $\frac{\text{Total transverse inertia}}{\text{Total spin inertia}}$ (Required for the Spin-Up Mode only)	More difficult to achieve as result of additional physical constraints and ratio of $\frac{\text{Total transverse inertia (rotor and stator)}}{\text{Rotor spin inertia only}}$ (Hinged weights required to attain this)

3. SYSTEM B - GENERAL DISCUSSION

This system requires the use of slip rings, bearings, and gears with consequently reduced reliability. Additional weights must be hinged out in order to provide the proper inertia ratio.

As a result of these additional constraints, a small bearing spread results. However, by proper design this effect can be minimized. A further effect is to shift the principal axis as a result of the product of inertia term due to antenna pivot at one end. This is necessary to meet inertial and TV transmission constraints. This effect does not have a serious influence, however, if this shift is kept below 20° to 30°. This system can be definitely implemented, although the additional constraints make it a more difficult task than System A.

F. DETAILED DISCUSSION OF SYSTEMS A AND B

1. SYSTEM A

a. Power Supply Consisting Solely of 15 ft^2 of Projected Solar Cell Area Plus Batteries (see Figure SK1720924)

(1) General Description

As explained previously, increased power requirements dictated an increase of the solar panel array to 15 ft^2. As a result it is not possible to constrain the payload vehicle to the basic Ranger shroud and bus and still meet the inertia ratio, c.g. and volume constraints. At best, 10 ft^2 of projected area is the largest solar array which can be accommodated by this standard design.

By utilizing a straight shroud in the vicinity of the payload capsule along with spreading the bus paddles, the 15 ft^2 projected solar cell area can be readily accommodated. Of course, this results in an additional weight penalty as a result of the heavier shroud weight, as well as frictional drag. Furthermore, the structural weight will be increased. However, a very flexible design can be obtained for adapting to changes in subsystem design.

The projected Atlas-Agena payload capability must be reviewed to insure that the all-solar array design of 15 ft^2 projected area can be readily accommodated.

The following description will be compared to the design originally proposed in the first study report.

(2) Payload Capsule Adapter

As indicated in the Phase I report, a Marman clamp with explosive bolts ties the capsule to the adapter secured to the top of the JPL bus. Ejection springs will be used to separate the payload capsule from the bus. The bottom thermal shield, for maintaining the retro-rocket temperature, has been retained.

(3) Retro-Rocket

The pre-retro spin-up rockets have been relocated to the outer diameter in order to decrease their weight. A single explosive bolt now secures the retro-rocket. The latter will be ejected, after combustion, to avoid unbalance due to unburned propellant. The ejection may require some guidance to preclude bumping the payload capsule. This may be accomplished by guide rods or an external guidance shroud.

(4) The Baseplate

This now supports components on both sides to obtain a suitable inertia ratio.

(5) The Structural Frame

Basically the same design as noted in the Phase I LOC study report. However, this frame will be heavier as a result of the larger loads associated with the increased frame width and inertia of the payload vehicles.

(6) The Bottom Thermal Shield

Basically the same design.

(7) The Main Thermal Shield

This may be eliminated, since sufficient heating may be provided by projecting fins. However, some form of protection may still be required to prevent deposition of products from the combustion of the retro-rocket.

(8) The Top Thermal Shield

Basically the same design.

(9) Omni-Directional Antenna

Basically the same design except this has been dropped to lower the c.g.

(10) Solar Panels

As explained previously, this has been increased to 15 sq. ft. as a result of increased power requirements. The panels now consist of 2 x 7 = 14 trapezoidal sections. The area facing the moon does not require solar cells; their contribution in this zone is negligible.

(11) Team Dampers

These have been moved out to increase the damping rate.

(12) The De-Spin Yo-Yo

(13) Attitude and Stabilization Jets

(14) Spin-Up Rockets

All three of these items have been increased in weight to accommodate the increased inertia.

(15) The Attitude and Stabilization Jet Fuel Tank

This has now been configured to a torus and has been dropped to lower the c.g. as well as providing stiffening for the structural frame.

(16) L-Band Directional Antenna

The antenna has been relocated so that its c.g. lies on the vehicle principal axis. The product of inertia term has been effectively cancelled regardless of directional antenna orientation. This has been accomplished by supporting the antenna with a gooseneck frame at the c.g. colinear with the vehicle principal axis.

(17) Vidicon and Horizon Scanners

These have now been separately mounted to the baseplate. However, a common locating plug will orient the vidicon with respect to the horizon scanner assembly.

(18) Thc Digital Solar Aspect Sensor

Basically the same design.

(19) Transmitter and Battery Assembly

Basically the same design, except for increased power requirements.

(20) Gyros and Other Electronic Components

Basically the same mounting.

b. A Power Supply Consisting Solely of a 75-watt RTG and Batteries (see Figure SK1720922)

This system permits the use of a standard Ranger shroud and bus without any redesign. The envelope and constraints imposed by the existing design are readily met. The general design is "disc-like" with a lower structural weight.

The baseplate which supports the electronic equipment is similar to the design submitted in the first LOC study report (Phase I). As in the previous design, the electronic units have been mounted on both the top and bottom surfaces of the baseplate. On the top surface, these units have been located between the RTG ribs. This design is not expected to present any problems. If necessary, the electronic units can be insulated. This area will be further investigated.

A Marman clamp is utilized here to secure the payload capsule to the retro-rocket which carries the load via the adapter assembly to the Ranger bus.

Aside from the expansion in diameter required by the all-solar cell array design, this design is basically the same.

It should be noted that the RTG permits a smaller battery pack as a result of permitting charging during the dark period.

This design presents a good packing factor and can readily adapt for variations in subsystem design.

c. A Combination Power Supply Consisting of a 30-watt RTG, 8.3 ft^2 of Projected Solar Cell Area and Batteries (see Figure SK1720923)

This combination results in a slight weight saving compared to the 75-watt RTG system, as far as the power supply is concerned. However, the additional structural weight required will offset this.

The electronic units do not "see" the thermal ribbing of the RTG. Consequently, the interface problems between these elements are simplified. In general, other design aspects are similar to the all-RTG system. The basic difference lies in meeting the c.g. and inertia-ratio constraints.

2. SYSTEM B

a. General Mechanical Problems

As explained previously, this system requires moving parts such as slip rings, bearings and gears with consequently reduced reliability. The stator and rotor will be locked during the launch and transfer modes. Relative rotation between the two inertial masses will be permitted after initial spin-down. Because of the additional mechanical complexity, it will be necessary to utilize pivoted weights to achieve the desired inertia ratio and c.g. restraints. This is discussed more fully in the next section.

A small bearing spread is necessary to meet the envelope constraint. To some extent this can be improved by using angular contact ball bearings suitably installed to increase the effective resisting moment arm. It should be pointed out that eccentricities in the bearing may be reduced to minimize the inertial unbalance. This can be accomplished by a combination of a slight preload, higher class bearing, and alignment of bearing high points and eccentricity match.

The bearings and slip rings introduce problems because of operation in a space environment. Although a development program for this particular design is required, these are within the present state-of-the-art.

Again, as a result of meeting the envelope constraints, with the retro-rocket on one side and the tape recorder and directional antenna on the other, very little space is provided for the rotor-stator drive. This imposes an additional restriction in meeting the design requirements.

A shift of principal axis occurs as a result of the product of inertia term resulting from antenna erection. However, this does not have a serious influence if this shift is kept below 20° to 30°.

This system can be implemented; however, inertial and spatial constraints make this a more difficult task than System A.

b. Inertia Ratio Problems

The inertia ratio of a preliminary System B configuration has been calculated to be 1.555 (rod-shaped). A study of the configuration and its constraints has shown that this inertia ratio cannot be made less than unity for this preliminary concept regardless of diameter changes. The concept has been to evolve the simplest design consistent with system constraints. Therefore, the preliminary design closely resembled the System A design presented in the Phase I report except for the rotating feature. However, this is no longer possible without unreasonable weight penalty, and a scheme to correct the inertia ratios is mandatory (See section on attitude control). The most promising approach appears in the extension of the mass of the rotating part to increase the inertia about the figure axis. A mass on a hinged arm can be accommodated with an extended radius of 48 inches. Four arms would be required for a total weight of approximately 14 pounds. By locating these weights at the top of the solar array and controlling the type of orbit, complete freedom from cell shadowing can be assured. Effects on the L-band dish can be minimized by selecting a non-metallic material for the arms. The masses of course can be any capsule payload items that lend themselves to the situation.

Stresses generated during erection of the arms while rotating will be controlled by two means: (1) by limiting the initial speed at time of erection to the minimum possible, which is the ratio of the erected spin inertia to the latched spin inertia times the final spin speed (10 rpm), and (2) by providing a dashpot hinge mechanism. The latter feature is also necessary to synchronize the opening action of all four arms. By connecting each double-acting dashpot to its neighbor in a series closed-loop fashion, equal displacement must occur at each arm. If this dashpot is a bellows and orifice, a completely sealed system is possible with zero leakage.

c. Detailed Description (see Figure SK1720955)

The basic difference in design here is the interface area between the rotor and stator.

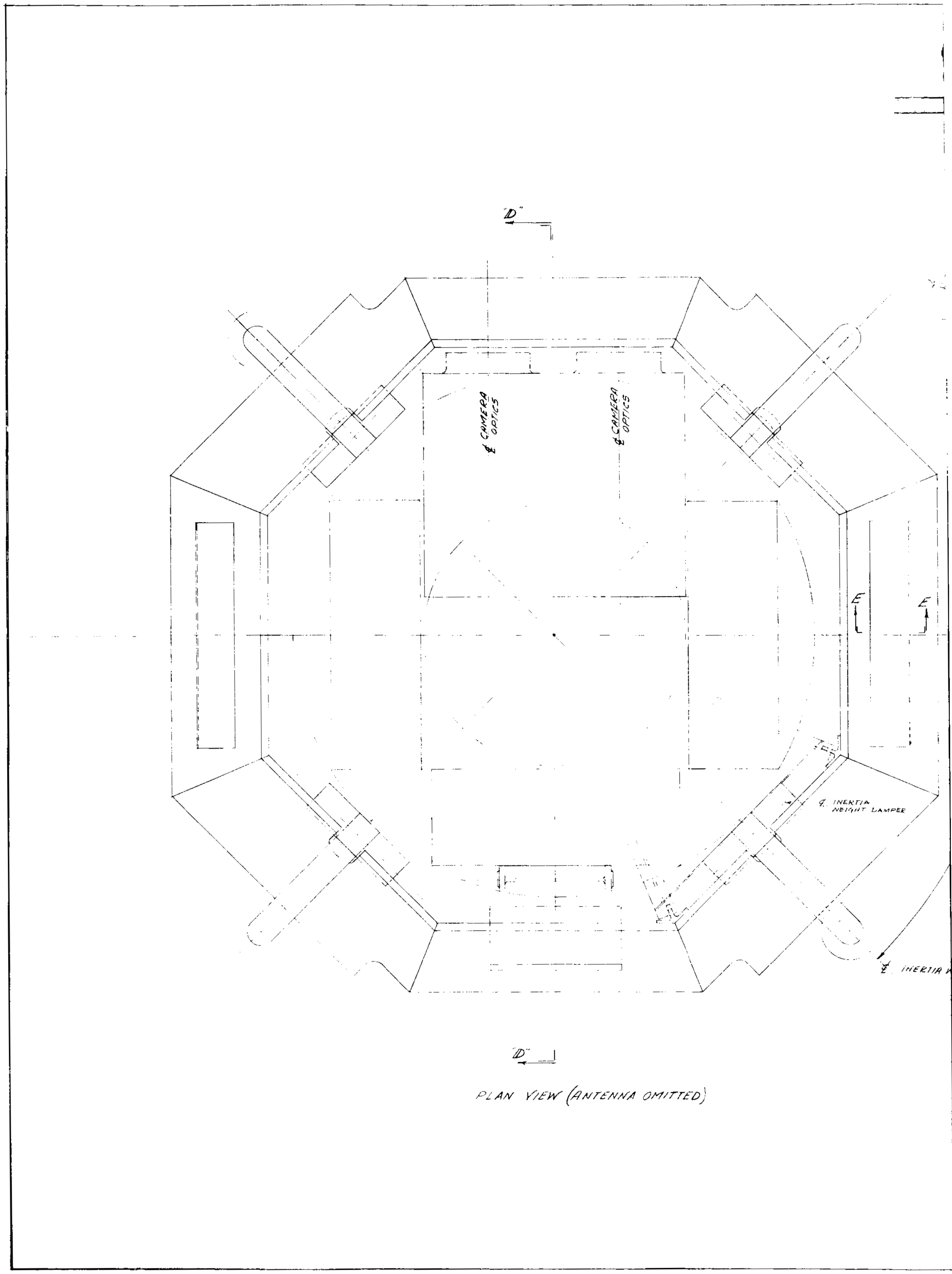
Torque tube bearings separate the rotor and stator. Bearing lubrication will be provided by maintaining the lubricant vapor pressure with evaporation from an oil fed wick. The use of a low vapor pressure oil and a face labyrinth seal will minimize oil evaporation loss.

The motor-reducer is located within the tube supporting the stator section. This section in turn supports the tape recorder, the horizon and scanner assembly and the directional-antenna.

The rotor includes privoted weights to provide the desired inertia ratio.

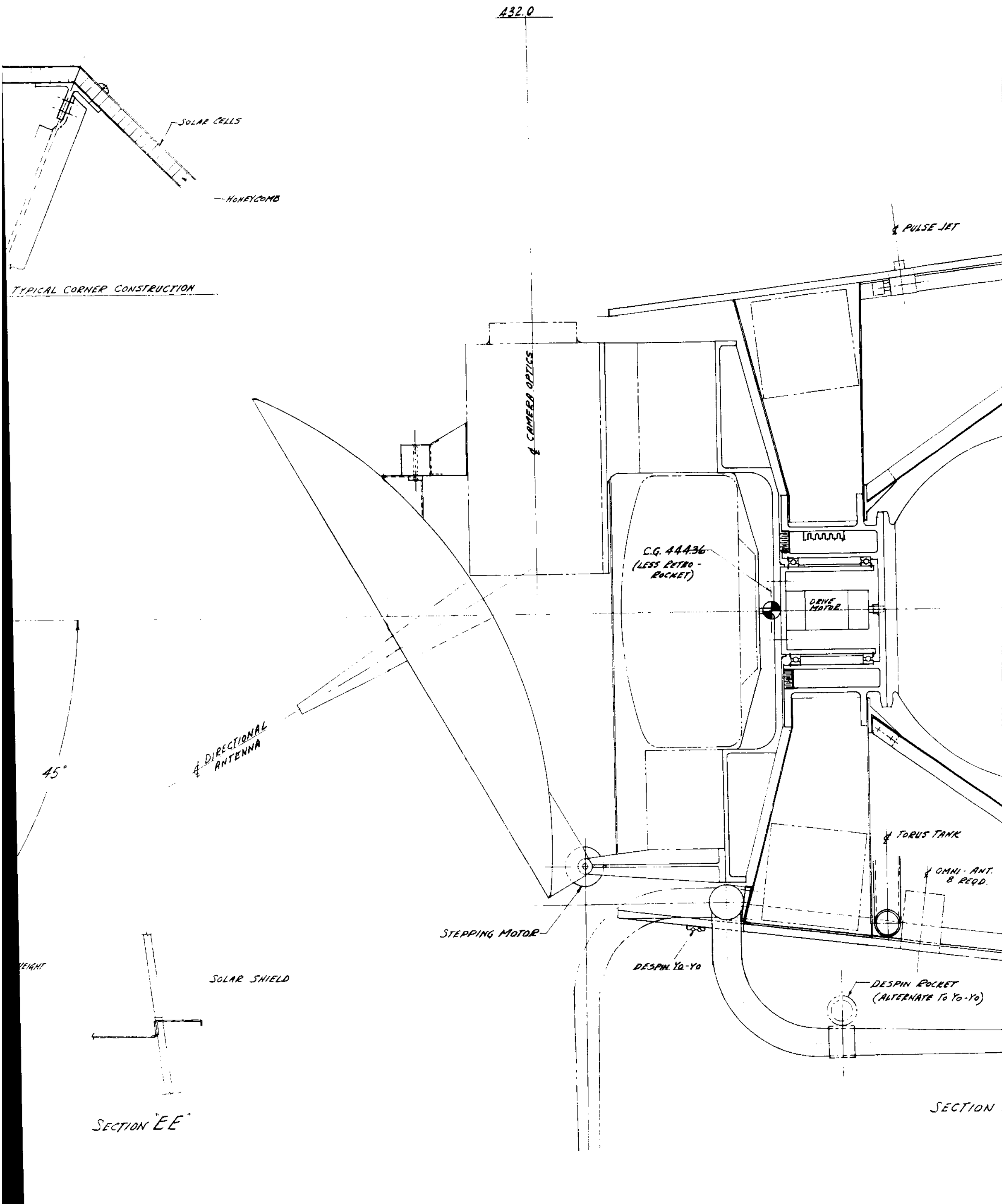
An additional latch is utilized to secure the rotor and stator during launch, spin-up, and de-spin modes. This virtually removes the bearing and gear tooth loads in these operational modes. The rotor is unlocked from the stator prior to achieving the inertially fixed position for the stator and final spin-up of 10 rpm for the rotor. To achieve compactness and reliability, the latch provided will be explosively actuated.

Other design areas are in general the same as the designs used in the "A" Systems.

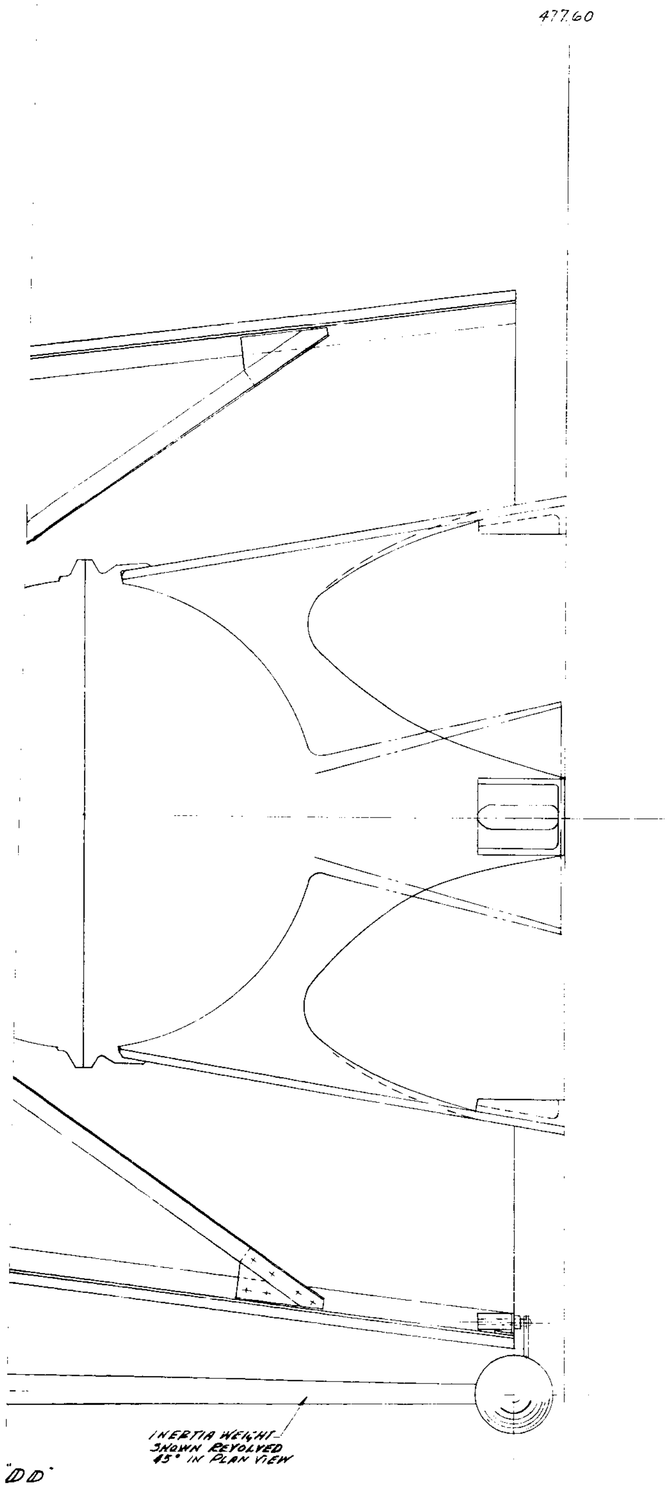


PLAN VIEW (ANTENNA OMITTED)

A ①


432.0
SOLAR CELLS
HONEYCOMB
TYPICAL CORNER CONSTRUCTION
℄ PULSE JET
℄ CAMERA OPTICS
C.G. 44.436
(LESS RETRO-ROCKET)
DRIVE MOTOR
℄ DIRECTIONAL ANTENNA
45°
℄ TORUS TANK
℄ OMNI-ANT. 8 REQD.
STEPPING MOTOR
DESPIN YO-YO
DESPIN ROCKET
(ALTERNATE TO YO-YO)
SOLAR SHIELD
SECTION "EE"
SECTION

A (2)



ITEM NO	COMPONENT	LOC.*	NO UNITS	UNIT WGT.	TOTAL WGT.	UNIT SIZE
1	CAMERA & OPTICS	F	2	13.25	16.50	7 DIA. X 13 LG.
2	CAMERA ELECTRONICS	F	2	5.70	11.40	430 CUBIC IN.
3	SYNC. GEN.	F	1	4.30	4.30	5.4 X 4 X 10.6
4	COMBINER-POWER SUPPLY	F	1	5.00	5.00	5.4 X 4 X 14.5
5	TAPE RECORDER	F	1	16.00	16.00	14 DIA. X 7
6	TRANSMITTER	F	1	12.00	12.00	430 CUBIC IN.
7	DIRECTIONAL ANTENNA	F	1	6.00	6.00	30 DIA. X 5 DEEP
8	TRANSPONDER	R	1	8.00	8.00	500 CUBIC IN.
9	OMNI-ANTENNA	R	8	1.25	10.00	2 X 3 X 7
10	COMMAND & DECODER	R	1	3.25	3.25	125 CUBIC IN.
11	TELEMETRY A	R	1	3.00	3.00	5 X 5 X 5
—	TELEMETRY B	F	1	1.00	1.00	4 X 4 X 5.5
12	TIMER	R	1	3.00	3.00	125 CUBIC IN.
13	HORIZON SCANNER	R	1	5.00	5.00	5 X 5 X 7
14	A PRECESSION GYRO	R	1	3.00	3.00	2 DIA. X 3.5 LG.
	B PITCH RATE GYRO	R	1	1.50	1.50	2 DIA. X 3.5 LG
15	TEAM DAMPERS	R	2	1.00	2.00	1 X 3 X 18
16	DESPIN YO YO	R	2	1.50	3.00	2 X 2 X 2
17	SPIN UP ROCKETS	R	2	1.00	2.00	.75 DIA. X 3.00 LG.
18	BATTERIES	R	2	12.50	25.00	5 X 5 X 16
19	BATTERY ELECT. PACK	R	1	6.00	6.00	5.00 X 5.00 X 5.00
20	SOLAR ARRAY	R	1	32.00	32.00	9 SQ. FT (PROJ. AREA)
21	PULSE TORQUER SYSTEM	R	1	7.50	7.50	1.50 DIA 30.0 DIA. TORUS
22	MARMON CLAMP	R	2	2.00	4.00	10.00 DIA. & 20.00 DIA.
23	CABLING	F&R	1	5.00	5.00	
24	DIPLEXER CABLE		1	1.25	1.25	
25	STRUCTURE MAIN					
	A BASEPLATE	F	1	15.00	15.00	28 DIA. X 3 HIGH
	B VANES & BULKHEADS	R	1	10.00	10.00	
	C ANGLES & TRUSS	R	1	5.00	5.00	
26	RETRO-ROCKET	F	1	188.1	188.1	20.00 DIA X 28.00
27	BRG. & SLIP RINGS	F/R	1	3.00	3.00	12.00 DIA. X 5 HIGH
28	TRANSITION SKIRTS	R	1	13.00	13.00	
29	HEAT SHIELD (TOP)	F	1	1.00	1.00	
30	HEAT SHIELD (FIN)	R	2	.25	.50	
31	CONNECTORS & UMBIL.	R	1	1.00	1.00	
32	CONTINGENCY	R/F	1	5.00	5.00	
33	MOTOR STEPPING	F	1	2.00	2.00	2.00 DIA X 2.00 LG.
34	CIRCULATOR	R	1	3.00	3.00	5 X 5 X 5.5
35	PITCH SERVO MOTOR	R/F	1	3.00	3.00	1.00 DIA X 3.6 LG.
36	INERTIA WEIGHT SYSTEM	R	1	25.00	25.00	
			TOTAL WEIGHT		476.30	

* R = ROTATING
F = FIXED

FIRST MADE FOR	
NEXT ASS'Y	USED ON

ALL EXTERNAL THREADS TO BE CLASS 2A BEFORE PLATING, AND CLASS 2 AFTER PLATING. ALL INTERNAL THREADS TO BE CLASS 2B, UNLESS OTHERWISE SPECIFIED.

DIMENSIONS ARE IN INCHES AND INCLUDE THICKNESS OF PLATING.

TOLERANCES ON FINISHED DIMENSIONS UNLESS OTHERWISE SPECIFIED

BASIC DIMENSIONS	2 PLACE DECIMALS	3 PLACE DECIMALS
UP TO 6	± .02	± .005
ABOVE 6 TO 24	± .03	± .010
ABOVE 24	± .06	± .015

ANGULAR DIMENSIONS ± ½°

SEE RCA PURCH SPEC FOR STOCK TOL
SEE RCA 96400 FOR MFG TOL SPEC
SEE RCA 96404 FOR WORKMANSHIP

QTY REQD 504	QTY REQD 503	QTY REQD 502	QTY REQD 501	ITEM NO	CODE IDENT	PART OR IDENTIFYING NO	NOMENCLATURE OR DESCRIPTION

LIST OF MATERIALS

CONTRACT NO

COMMODITY CODE

DRAWN DATE

CHECKED DATE

DESIGN ACTIVITY APPD. DATE

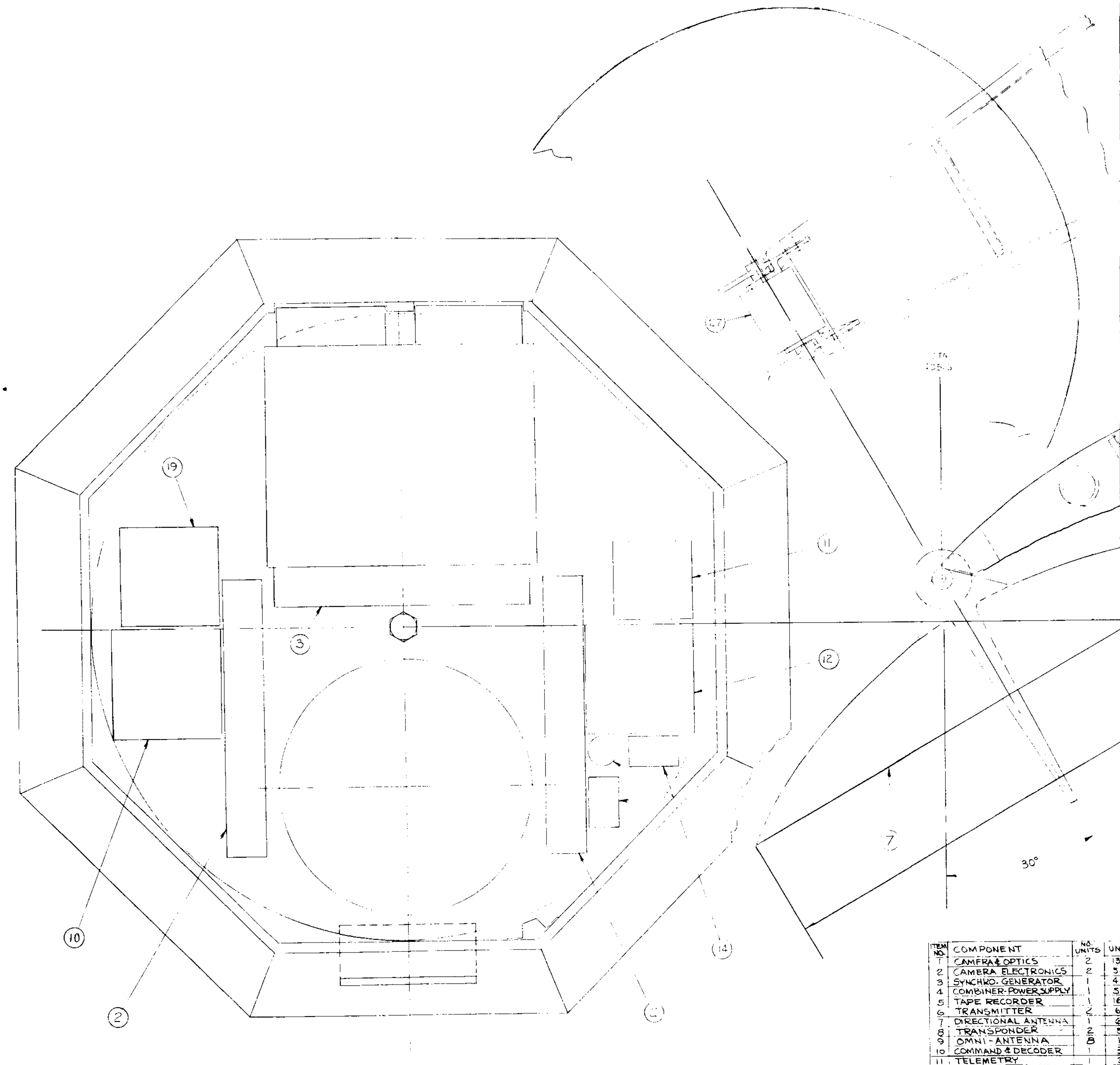
RADIO CORPORATION OF AMERICA
CAMDEN, N.J.
ASTRO-ELECTRONICS DIVISION PLANT

LUNAR ORBITER CAPSULE
ASSEMBLY
SPIN STABILIZED SYSTEM

CODE IDENT NO.	SIZE	
49671		SK 1720955
SCALE	WEIGHT	SHEET

A3

1

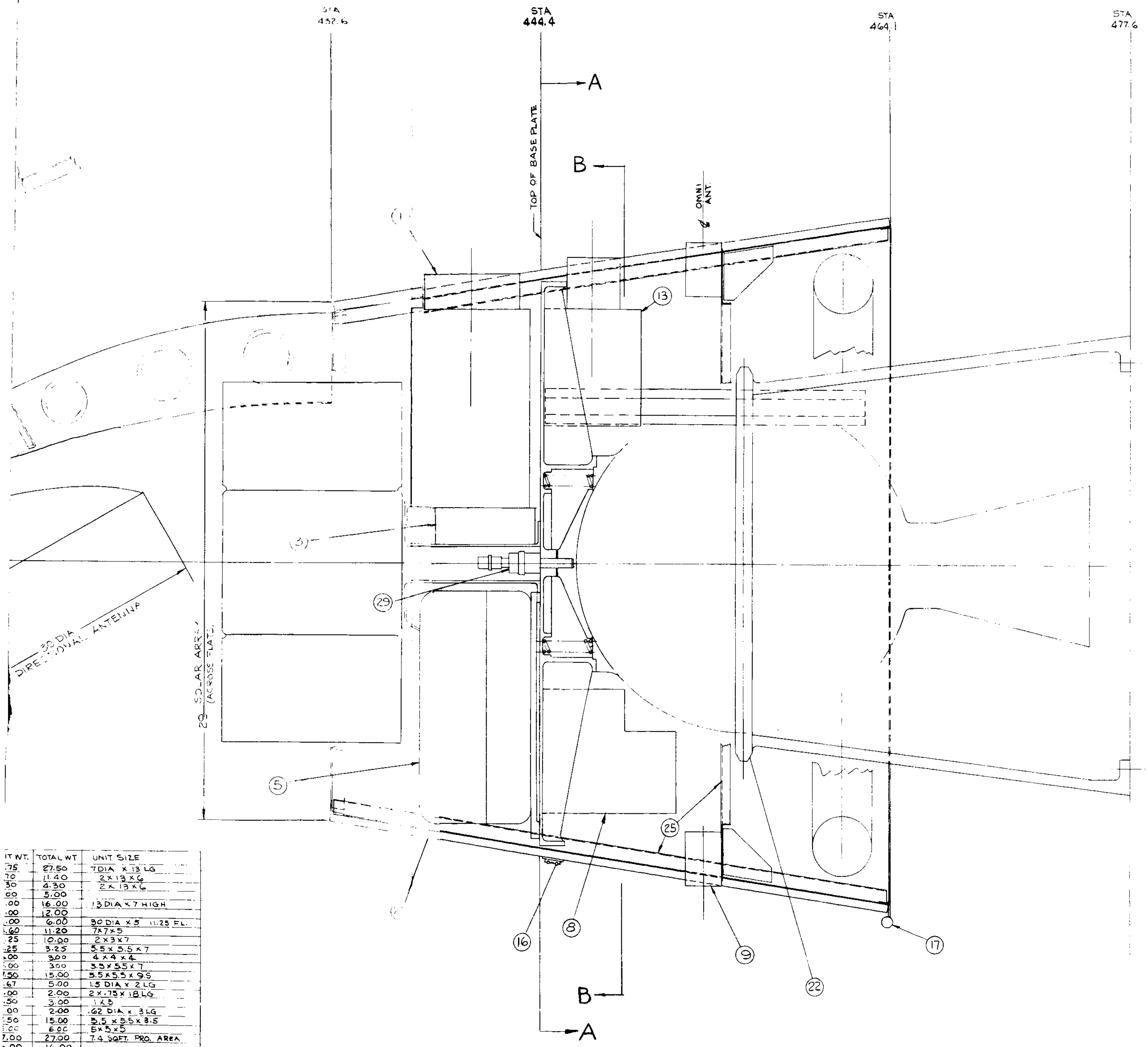


SEC A·A

ITEM NO.	COMPONENT	NO. UNITS	UN
1	CAMERA & OPTICS	2	13
2	CAMERA ELECTRONICS	2	5
3	SYNCHRO. GENERATOR	1	4
4	COMBINER-POWER SUPPLY	1	5
5	TAPE RECORDER	1	16
6	TRANSMITTER	2	6
7	DIRECTIONAL ANTENNA	1	6
8	TRANSPONDER	2	3
9	OMNI-ANTENNA	8	
10	COMMAND & DECODER	1	
11	TELEMETRY	1	
12	TIMER	1	
13	HORIZON SCANNER	2	
14	RATE GYROS	3	
15	TEAM DAMPERS	2	
16	DE-SPIN YO-YO	2	
17	SPIN-UP ROCKETS	2	
18	BATTERIES	2	
19	BATTERY PACK	1	
20	SOLAR ARRAY	1	2
21	PULSE TORQUER SYSTEM	1	
22	MARMAN CLAMP	1	
23	CABLING	1	
24	DIPLEXER CABLE	1	
25	STRUCTURE	1	
26	RETRO-ROCKET	1	
27	STEPPING MOTOR	1	
28	RADIOISOTOPE THERMOELEC GEN.	1	
29	EXPLOSIVE BOLT	1	

TO

B-(1)



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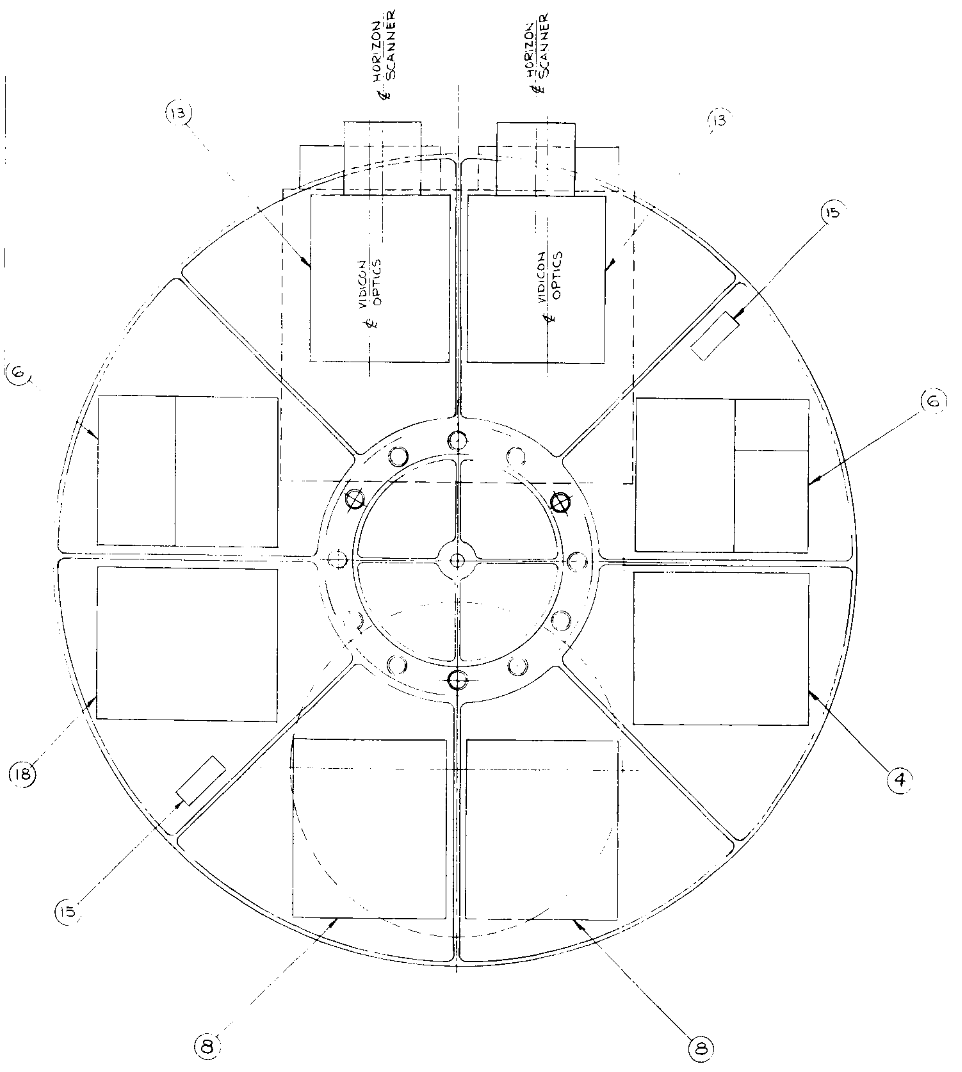
.00	16.00	13 DIA X 7 HIGH
.00	12.00	
.00	6.00	30 DIA X 5 11.25 F.L.
.60	11.20	7X7X5
.25	10.00	2X3X7
.25	3.25	5.5 X 5.5 X 7
.00	3.00	4 X 4 X 4
.00	3.00	5.5 X 5.5 X 7
.50	15.00	5.5 X 5.5 X 9.5
.67	5.00	1.5 DIA X 2 LG
.00	2.00	2 X .75 X 18 LG
.50	3.00	1 X 5
.00	2.00	.62 DIA X 3 LG
.50	15.00	5.5 X 5.5 X 3.5
.00	6.00	5X5X5
7.00	27.00	7.4 SQFT. PRO. AREA
.00	16.00	
.00	2.00	
5.00	5.00	
1.25	1.25	
5.00	35.00	
38.1	188.1	
00	2.00	3 DIA X 3.5 LG
5.00	25.00	20 DIA X 10 HIGH
1.00	1.00	

AL WT 463.00

B-2

REVISIONS				
SYM	ZONE	DESCRIPTION	DATE	APPROVED



SEC B-B

QTY REQD	QTY REQD	QTY REQD	QTY REQD	ITEM NO.	CODE IDENT	PART OR IDENTIFYING NO	NOMENCLATURE OR DESCRIPTION
504	503	502	501		LIST OF MATERIALS		

DIMENSIONS ARE IN INCHES AND INCLUDE THICKNESS OF PLATING

TOLERANCES ON FINISHED DIMENSIONS UNLESS OTHERWISE SPECIFIED

BASIC DIMENSIONS	2 PLACE DECIMALS	3 PLACE DECIMALS
UP TO 6	± .02	± .005
6 TO 24	± .03	± .010
ABOVE 24	± .06	± .015

ANGULAR DIMENSIONS ± 1/2°

SEE RCA PURCH SPEC FOR STOCK TOL
SEE RCA 98400 FOR MFG TOL SPEC
SEE RCA 98404 FOR WORKMANSHIP

FIRST MADE FOR	
NEXT ASS'Y	USED ON

ALL EXTERNAL THREADS TO BE CLASS 2A BEFORE PLATING, AND CLASS 2 AFTER PLATING. ALL INTERNAL THREADS TO BE CLASS 2B UNLESS OTHERWISE SPECIFIED.

CONTRACT NO

COMMODITY CODE

DRAWN B.Hutchuck DATE Sept 24 1962

CHECKED DATE

DESIGN ACTIVITY APPD. DATE

ISSUED Sept 25 1962

RADIO CORPORATION OF AMERICA
CAMDEN, N.J.
ASTRO ELECTRONIC DIV. PRINCETON N.J. PLANT

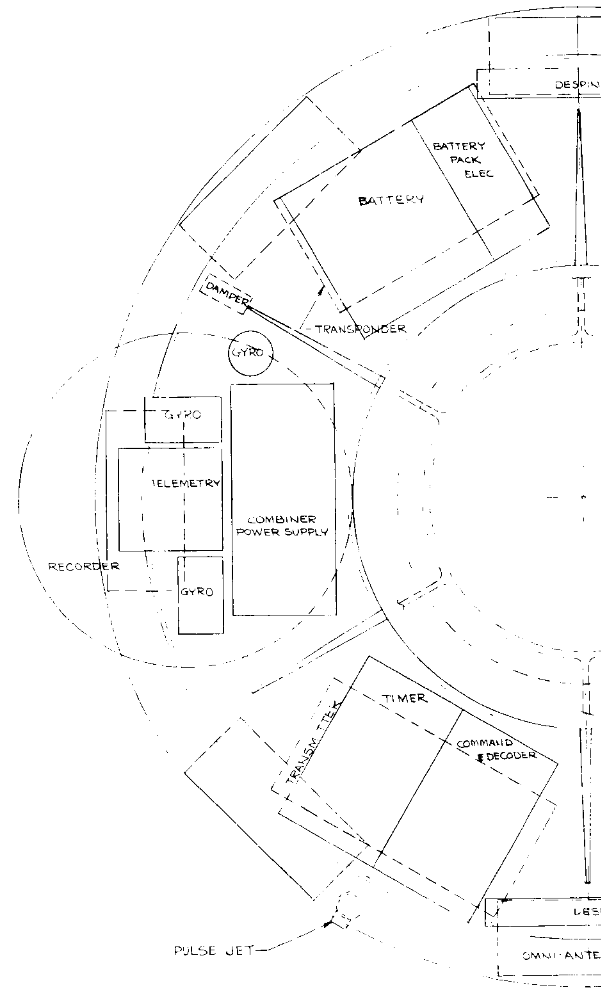
LUNAR ORBITER CAPSULE ASSEMBLY
3 AXIS GAS JET STABILIZATION SYSTEM
COMBINED SOLAR ARRAY & 25 WATT RTG

CODE IDENT NO 49671 SIZE SK 1720923

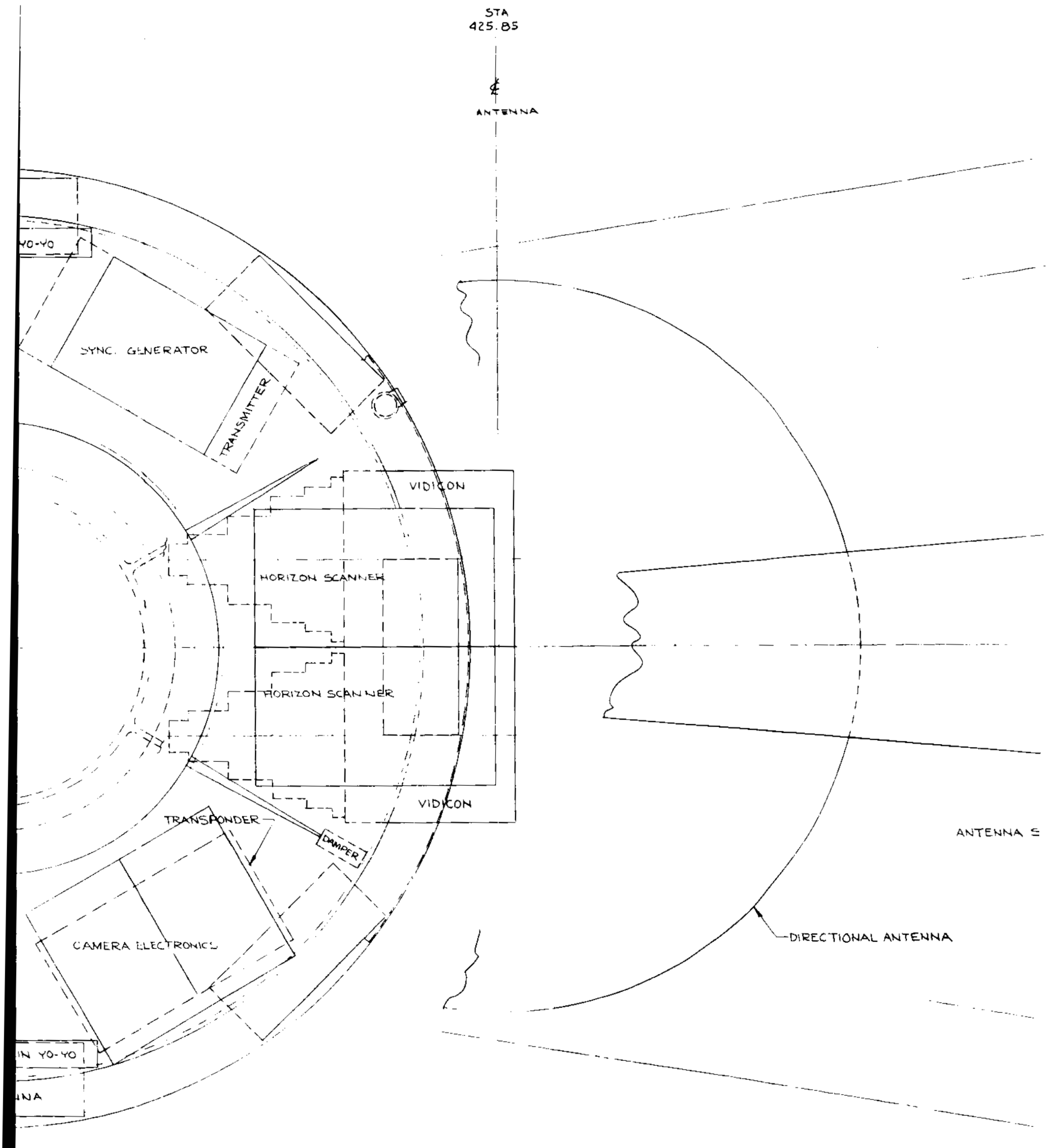
SCALE 1/2 WEIGHT SHEET 1 OF 1

B-3

ITEM NO	COMPONENT	NO UNITS	UNIT WT	TOTAL WT	UNIT SIZE
1	CAMERA & OPTICS	2	13.75	27.50	7 DIA × 13 LG
2	CAMERA ELECTRONICS	2	5.70	11.40	4.25 × 7 × 5
3	SYNCHRONIZING GENERATOR	1	4.30	4.30	4.25 × 7 × 5
4	COMBINER-POWER SUPPLY	1	5.00	5.00	4 × 9 × 4.5
5	TAPE RECORDER	1	16.00	16.00	13 DIA × 7 HIGH
6	TRANSMITTER	2	6.00	12.00	5 × 10 × 5
7	DIRECTIONAL ANTENNA	1	6.00	6.00	30 DIA × 5 - 11.25 F.L.
8	TRANSPONDER	2	5.60	[illegible]	5 × 10 × 5
9	OMNI-ANTENNA	8	1.25	10.00	2 × 3 × 7
10	COMMAND & DECODER	1	3.25	3.25	4.25 × 7 × 5
11	TELEMETRY	1	3.00	3.00	4 × 4 × 4
12	TIMER	1	3.00	3.00	4.25 × 7 × 5
13	HORIZON SCANNER	2	7.50	15.00	5.5 × 9.5 × 5.5
14	RATE GYRO	3	1.67	5.00	1.5 DIA × 2 LONG
15	TEAM DAMPER	2	1.00	2.00	2 × .75 × 18
16	DE-SPIN YO-YO	2	1.50	3.00	1 × 5
17	SPIN-UP ROCKET	2	1.00	[illegible]	.62 DIA × 3 LG
18	BATTERY	2	5.00	10.00	6 × 6.50 × 6
19	RETRO ROCKET	1	188.10	188.10	
20	PULSE TORQUE SYSTEM	1	[illegible]	16.00	
21	MARMAN CLAMP	2	2.00	4.00	
23	CABLING	1	[illegible]	5.00	
24	DIPLEXER CABLE	1	1.25	1.25	
25	STRUCTURE	1	25.00	25.00	
26	BATTERY PACK ELEC	1	6.00	6.00	2.50 × 6.50 × 6.0
27	RADIOISOTOPE THERMOELEC GEN	1	60.00	60.00	
28	STEPPING MOTOR	1	2.00	2.00	
			TOTAL WT	463.00	

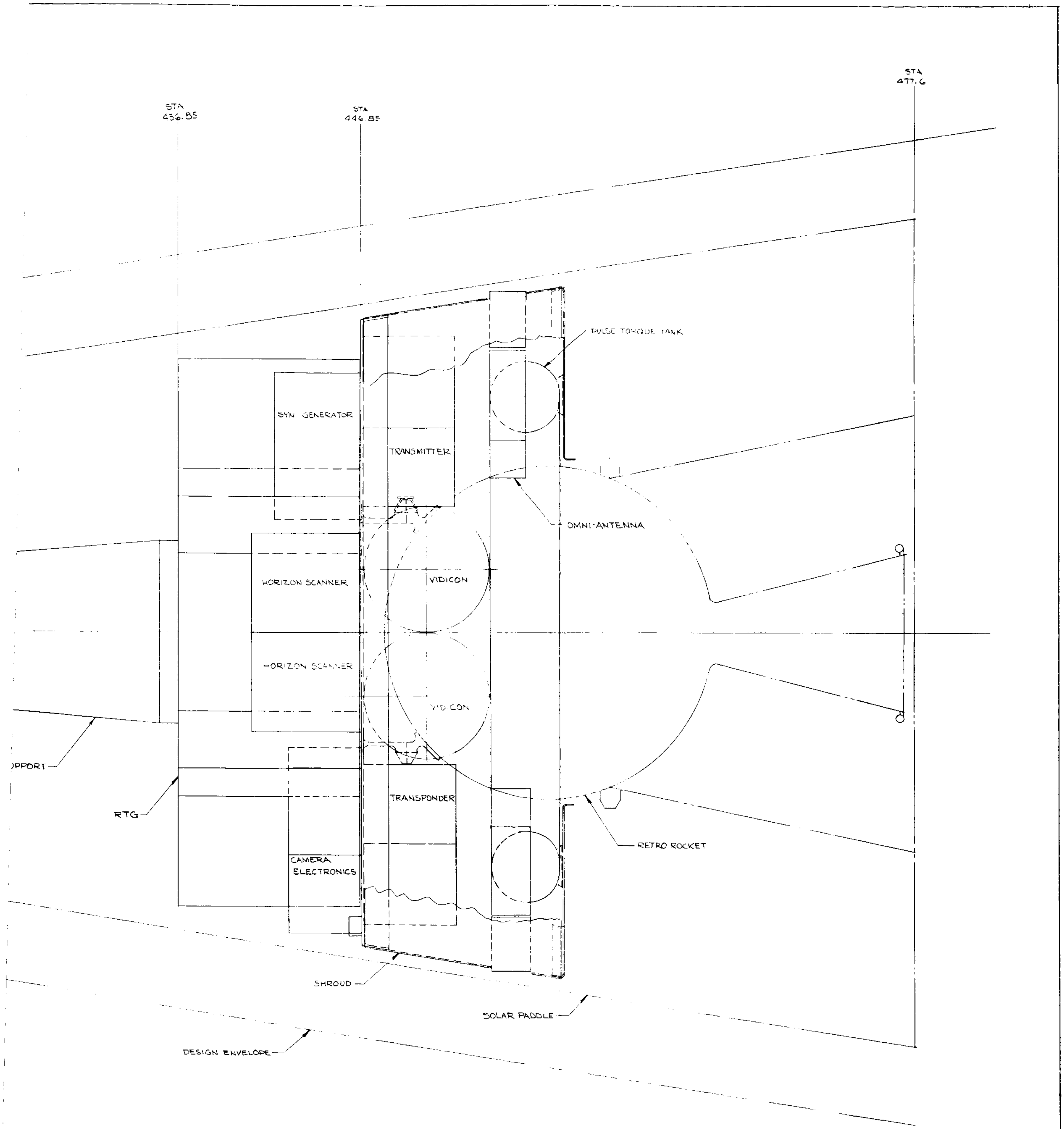


C-①



STA
425.85
ANTENNA
YO-YO
SYNC. GENERATOR
TRANSMITTER
VIDICON
HORIZON SCANNER
HORIZON SCANNER
VIDICON
TRANSPONDER
DAMPER
CAMERA ELECTRONICS
IN YO-YO
NNA
ANTENNA S
DIRECTIONAL ANTENNA

C



QTY REQD 504	QTY REQD 503	QTY REQD 502	QTY REQD 501	ITEM NO	CODE IDENT	PART OR IDENTIFYING NO	NOMENCLATURE OR DESCRIPTION
					LIST OF MATERIALS		

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FIRST MADE FOR	
NEXT ASS'Y	USED ON

ALL EXTERNAL THREADS TO BE CLASS 2A BEFORE PLATING, AND CLASS 2 AFTER PLATING. ALL INTERNAL THREADS TO BE CLASS 2B, UNLESS OTHERWISE SPECIFIED.

DIMENSIONS ARE IN INCHES AND INCLUDE THICKNESS OF PLATING

TOLERANCES ON FINISHED DIMENSIONS UNLESS OTHERWISE SPECIFIED

BASIC DIMENSIONS	2 PLACE DECIMALS	3 PLACE DECIMALS
UP TO 6	± .02	± .005
6 TO 24	± .03	± .010
ABOVE 24	± .06	± .015

ANGULAR DIMENSIONS ± ½°

SEE RCA PURCH SPEC FOR STOCK TOL
SEE RCA 96400 FOR MFG TOL SPEC
SEE RCA 96404 FOR WORKMANSHIP

CONTRACT NO.

COMMODITY CODE

DRAWN [illegible] DATE sept 20 1962

CHECKED DATE

DESIGN ACTIVITY APPD. DATE

issued Sept 25 1962

RADIO CORPORATION OF AMERICA
CAMDEN, N.J.
ASTRO ELECTRONIC DIV., PRINCETON N.J. PLANT

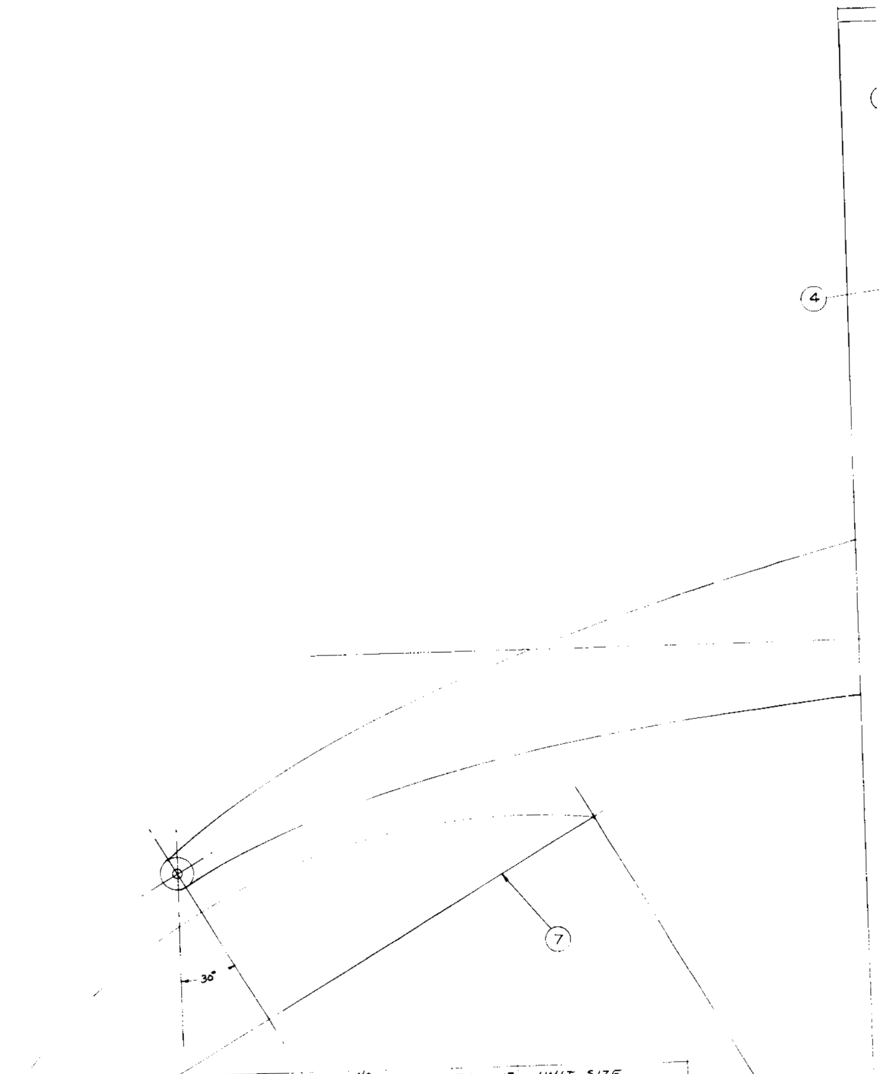
LUNAR ORBITER CAPSULE ASSEMBLY
THREE AXIS, GAS JET STABILIZED
75 WATT RTG POWER UNIT

CODE IDENT NO. 49671 | SIZE | SK 1720922

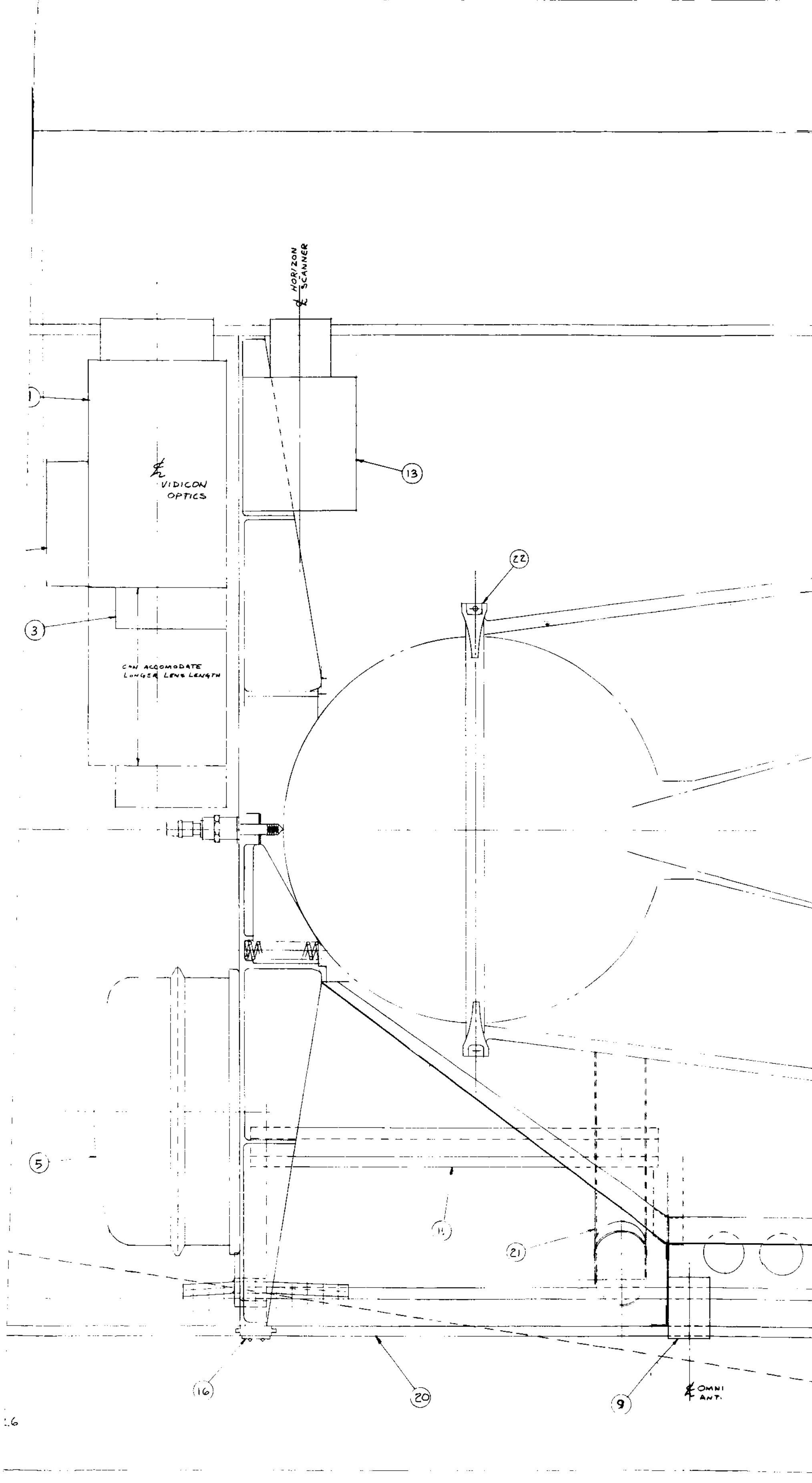
SCALE | WEIGHT | SHEET 1

(2)
C

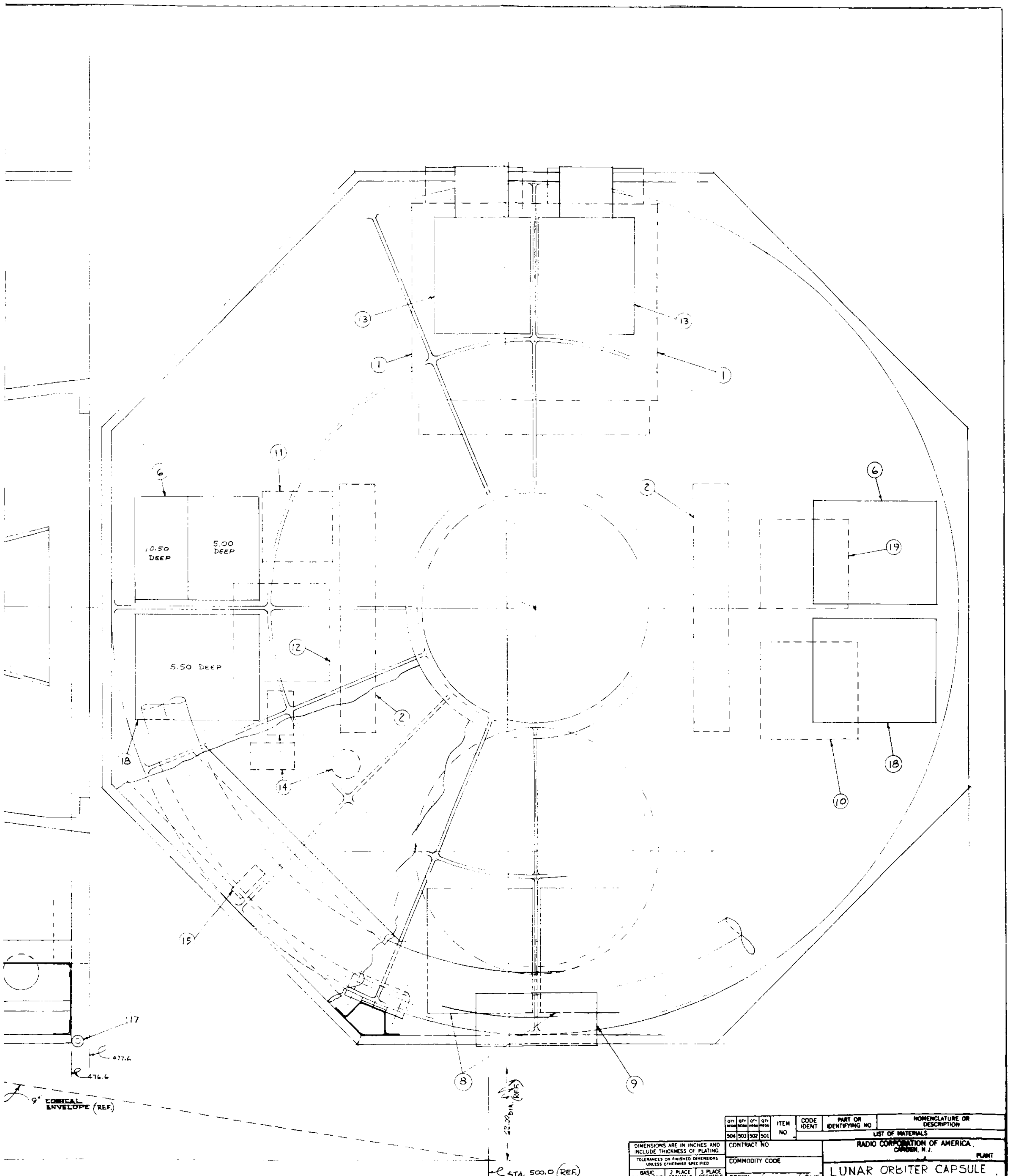
C - (3)



ITEM No.	COMPONENT	No. UNITS	UNIT WT.	TOTAL WT.	UNIT SIZE
1	CAMERA & OPTICS	2	13.75	27.50	7 DIA X 13 LG.
2	CAMERA ELECTRONICS	2	5.70	11.40	2 X 13 X 6
3	SYNC. GEN.	1	4.30	4.30	2 X 13 X 6
4	COMBINER - POWER SUPPLY	1	5.00	5.00	2 X 13 X 6
5	TAPE RECORDER	1	16.00	16.00	14 DIA. X 7 HIGH
6	TRANSMITTER	2	6.00	12.00	450 CUBIC INCHES
7	DIRECTIONAL ANT.	1	6.00	6.00	30 DIA. X 5 11.25 F.L.
8	TRANSPONDER	2	5.60	11.20	7 X 7 X 5
9	OMNI-ANT	8	1.25	10.00	2 X 3 X 7
10	COMMAND & DECODER	1	3.25	3.25	5.50 X 5.5 X 7
11	TELEMETRY	1	3.00	3.00	4 X 4 X 4
12	TIMER	1	3.00	3.00	5.5 X 5.5 X 7
13	HORIZON SCANNER	2	7.50	15.00	5.5 X 5.5 X 9.5
14	RATE GYROS	3	1.67	5.00	1.5 DIA X 2 LG.
15	TEAM DAMPERS	2	1.00	2.00	2 X .75 X 18 LG.
16	DE-SPIN YO-YO	2	1.50	3.00	1 X 5
17	SPIN UP ROCKETS	2	1.00	2.00	.62 DIA. X 3 LG
18	BATTERIES	4	6.25	25.00	5.5 X 5.5 X 3.5
19	BATTERY PACK	1	10.00	10.00	5 X 5 X 5
20	SOLAR ARRAY	1	52.00	52.00	16 SQ.FT. PROJECTED AREA
21	PULSE TORQUER SYSTEM	1	16.00	16.00	TORUS TANK & NOZZLES
22	MARMAN CLAMP	1	2.00	2.00	
23	CABLING	1	5.00	5.00	
24	DIPLEXER CABLE	1	1.25	1.25	
25	STRUCTURES	1	50.00	50.00	
26	RETRO-ROCKET	1	188.1	188.1	
			TOTAL WT.	489.0	


HORIZON SCANNER
VIDICON OPTICS
CAN ACCOMODATE LONGER LENS LENGTH
OMNI ANT.
1
3
5
9
13
16
20
21
22

D-2



QTY REQD 504	QTY REQD 503	QTY REQD 502	QTY REQD 501	ITEM NO.	CODE IDENT	PART OR IDENTIFYING NO	NOMENCLATURE OR DESCRIPTION

LIST OF MATERIALS

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FIRST MADE FOR	
NEXT ASS'Y	USED ON

ALL EXTERNAL THREADS TO BE CLASS 2A BEFORE PLATING AND CLASS 2 AFTER PLATING. ALL INTERNAL THREADS TO BE CLASS 2B UNLESS OTHERWISE SPECIFIED.

DIMENSIONS ARE IN INCHES AND INCLUDE THICKNESS OF PLATING.

TOLERANCES ON FINISHED DIMENSIONS UNLESS OTHERWISE SPECIFIED

BASIC DIMENSIONS	2 PLACE DECIMALS	3 PLACE DECIMALS
UP TO 6	± .02	± .005
6 TO 24	± .03	± .010
ABOVE 24	± .06	± .015

ANGULAR DIMENSIONS ± ½°

SEE RCA PURCH SPEC FOR STOCK TOL
SEE RCA 98400 FOR MFG TOL SPEC
SEE RCA 98404 FOR WORKMANSHIP

CONTRACT NO

COMMODITY CODE

DRAWN [illegible] DATE

CHECKED DATE

DESIGN ACTIVITY APPD. DATE

RADIO CORPORATION OF AMERICA, CAMDEN, N.J.

PLANT

LUNAR ORBITER CAPSULE
3 AXIS GAS JET STAB. SYSTEM
SOLAR CELL ARRAY POWER UNIT

CODE IDENT NO.	SIZE	
49671	E	SK 1720924
SCALE 1/2	WEIGHT	

D 3

SECTION IV

TELECOMMUNICATIONS SUBSYSTEMS

The following communications subsystems are required for the Lunar Orbiting Capsule:

(1) TV Transmission

(2) Tape storage

(3) Tracking transponder

(4) Command

(5) Timing

(6) Telemetry.

It is anticipated that systems requiring little or no development, and yet which offer the most in compatability with the Ranger facilities and operations will be used. The TV transmitter will be a low-power version of that used in Ranger 6 through 9. The tape recorder will be a modified Nimbus AVCS. The transponder will be the same as that used in Ranger 6 through 9. The command concept will also be the same as Ranger, although new hardware will be required for the actual programmer. The same is true for the telemetry subsystem. There will be two antennas: a dish and an "onmi".

The TV mapping of the lunar surface will require both direct picture taking and transmission as well as remote operation. For remote operation the tape recorder will be used to store the video information for later playback through the transmitter when the dish antenna is in view of the earth.

The recommended TV transmission subsystem makes use of many of the components used in the Ranger 6 through 9 TV payload. A single 20-watt transmission channel is used. A worst-case RF link analysis shows that a 20-watt, 960 Mc transmitter operating with a 30-inch dish will result in a 30-db peak-to-peak/rms signal-to-noise ratio, in a 60-kc baseband, with 8.1 db margin. This assumes the DSIF will contain a frequency feedback demodulator (FMFB), which is recommended.

Thus, the Ranger 6 through 9 transmitter can be operated at 20 watts. This will eliminate one intermediate power amplifier (IPA) and reduce the power requirements. The power amplifier (PA) will remain pressurized, as in Ranger 6 through 9, until it can be shown that an unpressurized PA can operate in the environment.

Engineering telemetry for the TV system will be multiplexed with the video information, in a manner similar to Ranger 6 through 9.

The tape recorder recommended for LOC is a modified Nimbus AVCS recorder, details of which are presented in Table 5. The recorder has a 60-kc bandwidth at a tape speed of 30 inches per second. The video is recorded by frequency-modulating a subcarrier oscillator. For transmission the tape is played back at the same speed into an FM demodulator.

The tape recorder will record for eight minutes in one direction and is capable of start-stop operation. The record time can be increased by using multiple tracks.

A 30-inch dish antenna with a circularly polarized feed is recommended for TV transmission. The dish will be hinged in elevation so that the boresight axis may be steered by command as the orbital plane precesses. The antenna beamwidth will be 30 degrees at the half-power points with a maximum gain of 15 db.

A two-way doppler transponder will be on the capsule for use by the DSIF for doppler and angle tracking of the orbiter. The transponder will also serve as the command receiver and telemetry transmitter. The Ranger L-band, mod II, transponder as used in the Ranger bus is recommended. The high power, 3-watt, power amplifier is required because of the relatively wideband attitude control telemetry requirements. The transponder will operate with the omni antenna.

The omni antenna will consist of an array of eight half-wave slot radiators. Each slot is excited by a cavity. The eight cavities are fed equal, in-phase signals by using seven power-dividers. The azimuth beamwidth is 360 degrees and the elevation beamwidth is 120 degrees. The antenna gain is -3 db and is linearly polarized.

The telemetry system recommended for LOC is similar to that used in the Ranger 6 through 9 bus, so that the present DSIF data reduction methods and equipment may be used. The number of telemetry channels can be reduced to conserve weight and power because of the relatively long mission time compared to a Ranger impact mission. Although the detailed requirements for telemetry data are not known at this time, it is safe to assume that except for the attitude control sensors all the data rates are very slow, and therefore slow commutating rates may be used.

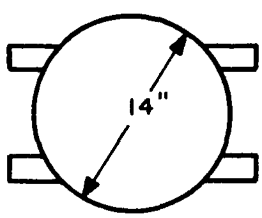
The command subsystem recommended for LOC is that used in the Ranger 6 through 9 bus, so that LOC is compatible with Ranger DSIF command procedures. The Command detector and decoder will be the same as Ranger, but the CC&S function will require a new design. The detailed requirements of each of the subsystems requiring command is not known at present; for that reason details of the command sequencer have not been worked out.

There will be a central timing source to be used by all subsystems for sequenced operations, such as remote picture taking.

Table 6 presents a summary of weight and power requirements for the telecommunications subsystems.

TABLE 5. NIMBUS AVCS TAPE RECORDER

Tape, one mil mylar base		1/2 inch wide
Length of tape		1200 feet
Tape speed		30 inches/sec.
Records in one direction for 8 minutes		
Plays back in opposite direction for 8 minutes		
Starting time		Less than 1 second
Stopping time		Less than 2 seconds (dynamic brake used)
Voltage input		-24.5 volts DC
Power input (initial surge for one second)		15 watts
Power input (steady state)		10 watts
Channels		4 total
Video	3 (72 kc to 120 kc -6.5 VDC into -11.5 VDC)	
Timing	1 (50 kc 50% AM)	
Erase		DC Permanent Magnet
Enclosure Size		8 inches high



Weight		17 pounds
Angular momentum compensated to ± 0.02 lb.-in.-sec.		
Impedances:	Power Converter Input	2,000 ohms
	Modulator Input	40,000 ohms
	Timing Input	620 ohms
	All Channel Outputs	1,500 ohms

Output Levels - All channels unloaded 6 Volts (p-p)

Wow and flutter - Measured values N12 Recorder 3-19-62

Frequency (cps)	Wow & Flutter % rms
0.5 to 30	0.015
30 to 300	0.04
300 to 5000	0.085
DC to 5000	0.11

The above do not represent spec limits, but are typical values.

TABLE 6. SUMMARY OF WEIGHTS AND POWER FOR THE TELECOMMUNICATIONS SUBSYSTEMS

Subsystem	Weight (pounds)	Power (watts)
TV Communications		(unregulated DC)
Transmitter and power supply	12.0	140
Tape recorder	16.0	10 (15 start)
High-gain antenna	6.0	10 (intermittent)
Tracking, Telemetry and Camera		(regulated DC)
Transponder and power supply	7.95	26
Diplexer	1.25	
Circulator	3.0	
Omni-antenna	10.0	
Command detector and decoder	3.24	2.5
Timing	3.0	3.0
Telemetry	4.0	2.0
Total	66.44 pounds	

SECTION V

TELEVISION SUBSYSTEM

A. INTRODUCTION

A block diagram of the television subsystem is shown in Figure 6. Two cameras are provided, and each is associated with a camera electronics unit. The cameras are exposed alternately at intervals of 6 seconds, so that the total elapsed time between successive exposures of a given camera is 12 seconds. The video signals provided by the cameras are added together in a combiner unit. The stored image is read out over a period of 6 seconds alternately from each of the cameras. One camera is erased and prepared for its next exposure while the other is being read.

The timing and synchronizing signals associated with operation of the camera are provided by a synchronizing generator shown in Figure 6. These signals are routed to the unit marked "Combiner Signal Distribution and Power Supply," where they are utilized as required by the combiner section and distributed to the two cameras by way of their corresponding camera electronics units.

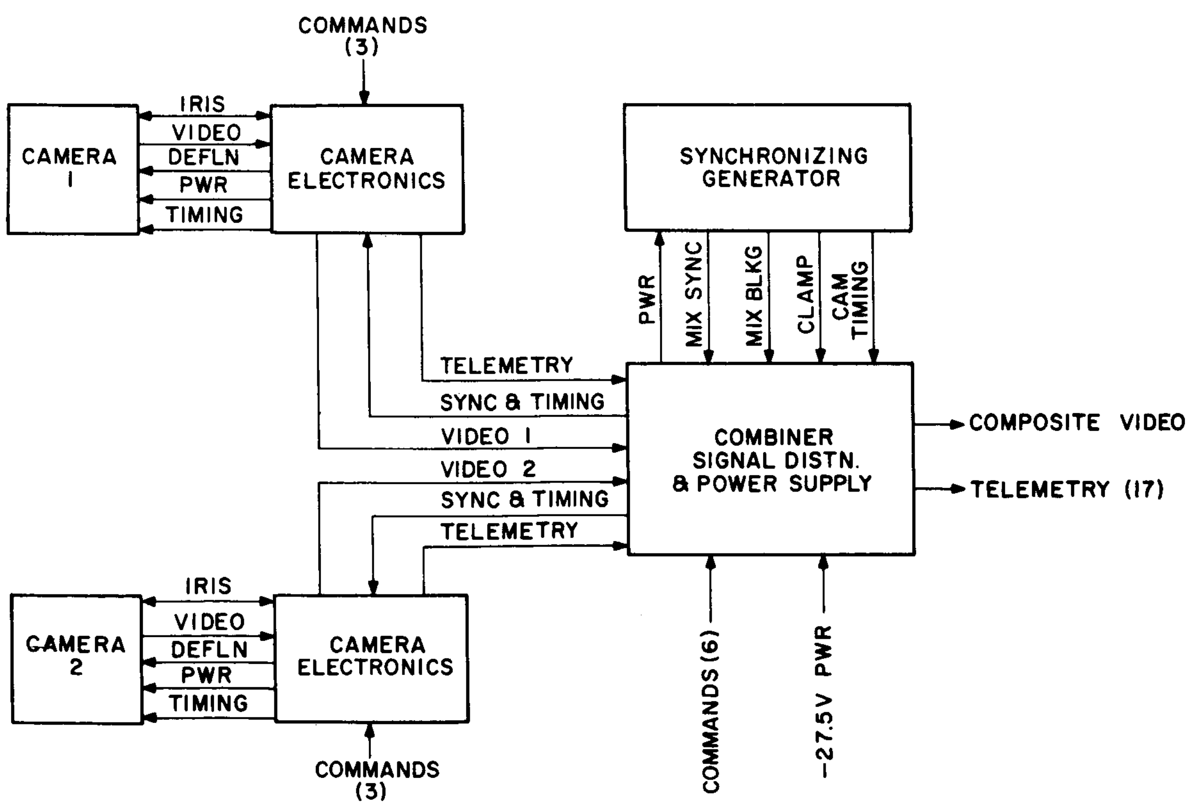


Figure 6. Block Diagram of the Television Subsystem

The Camera Electronics unit provides the necessary power, timing and deflection signals to its associated camera, accepting the basic timing pulses from the synchronizing generator and providing the appropriate amplifying or waveshaping circuits for this purpose. It also serves to control the camera iris in response to commands from the command receiver by furnishing the necessary servo circuits for this control. In addition, it serves to deliver telemetry signals to the signal distribution circuits.

Each camera contains a vidicon camera tube and associated focus-deflection coil assembly, together with the optical system including the iris and iris drive mechanism, and the shutter.

B. SYSTEM SPECIFICATIONS

The following tabulation shows the major parameters of the system:

Number of scanning lines	1024
Frame time (V)	6 sec
Vertical deflection frequency (read)	1/6 cps
Vertical deflection frequency (erase)	7281-7/9 cps
Horizontal deflection frequency	170-2/3 cps
Horizontal period (H)	5.859 msec
Horizontal blanking (0.05 H)	0.293 msec
Vertical blanking (0.015 V)	90 ms
Baseband	62.5 kc
Vertical Resolution (nominal)	700 lines
Horizontal Resolution	696 lines

The deflection frequencies are based on the use of straight binary counters in order to simplify the required counter chain. With the blanking percentages chosen and the baseband available (62.5 kc), the frame time is calculated to be 6 seconds. A total of 1024 scanning lines requires a count of 2^{10} between vertical and horizontal deflection frequencies, leading to the value shown (170-2/3 cps) for the horizontal deflection frequency.

It is to be noted that the vertical deflection frequency is changed during the erase cycle, being increased until it is higher by a factor of 42-2/3 than the horizontal deflection frequency, which remains unaltered. We have found that this approach materially improves the erasure. The reason for adopting the numerical ratio of 42-2/3 is because it provides a triple interlace of the erasure scanning and should effectively prevent the formation of erasure bars due to fortuitous phase lock occuring between the horizontal and vertical frequencies, which can occur when the erase vertical frequency is permitted to run free. The factor is easily realized in a frequency divider circuit having a division ratio of 3, as will be discussed later.

C. SYNCHRONIZING GENERATOR

A block diagram of counter circuits of the generator is shown in Figure 7. It is seen that there are four basic output frequencies, namely:

f_o... the basic frequency (1/12 cps) used to time the expose and erase cycles of each camera

f_V... the vertical deflection frequency (1/6 cps) used during the read cycle

f_H... the horizontal deflection frequency used during both read and erase cycles

f_V'... the vertical frequency used during the erase cycle.

These basic waveforms are processed in waveshaping circuits to provide the required pulses of these frequencies for use in the system. The following pulse systems are derived from the above basic frequencies:

H Sync	for Camera Electronics Unit No. 1
V Drive	for Camera Electronics Unit No. 1
V' Drive	for Camera Electronics Unit No. 1
H Sync	for Camera Electronics Unit No. 2
V Drive	for Camera Electronics Unit No. 2
V' Drive	for Camera Electronics Unit No. 2
Shutter pulse	for Camera No. 1
Erase pulse	for Camera No. 1
Camera gate pulse	for Camera No. 1
Shutter pulse	for Camera No. 2
Erase pulse	for Camera No. 2
Camera gate pulse	for Camera No. 2
Clamp pulse	for Combiner Unit
Mixed Sync	for Combiner Unit
Mixed blanking	for Combiner Unit

Of these signals, the camera gate pulse is used to alternate between the V drive and V' drive for a given camera, thus placing it alternately in the read and erase modes. The camera gate pulse for one camera is 180° out of phase with that for the other, in order to permit the desired staggered timing relation between read and erase. This is shown in the basic camera timing diagram, Figure 8.

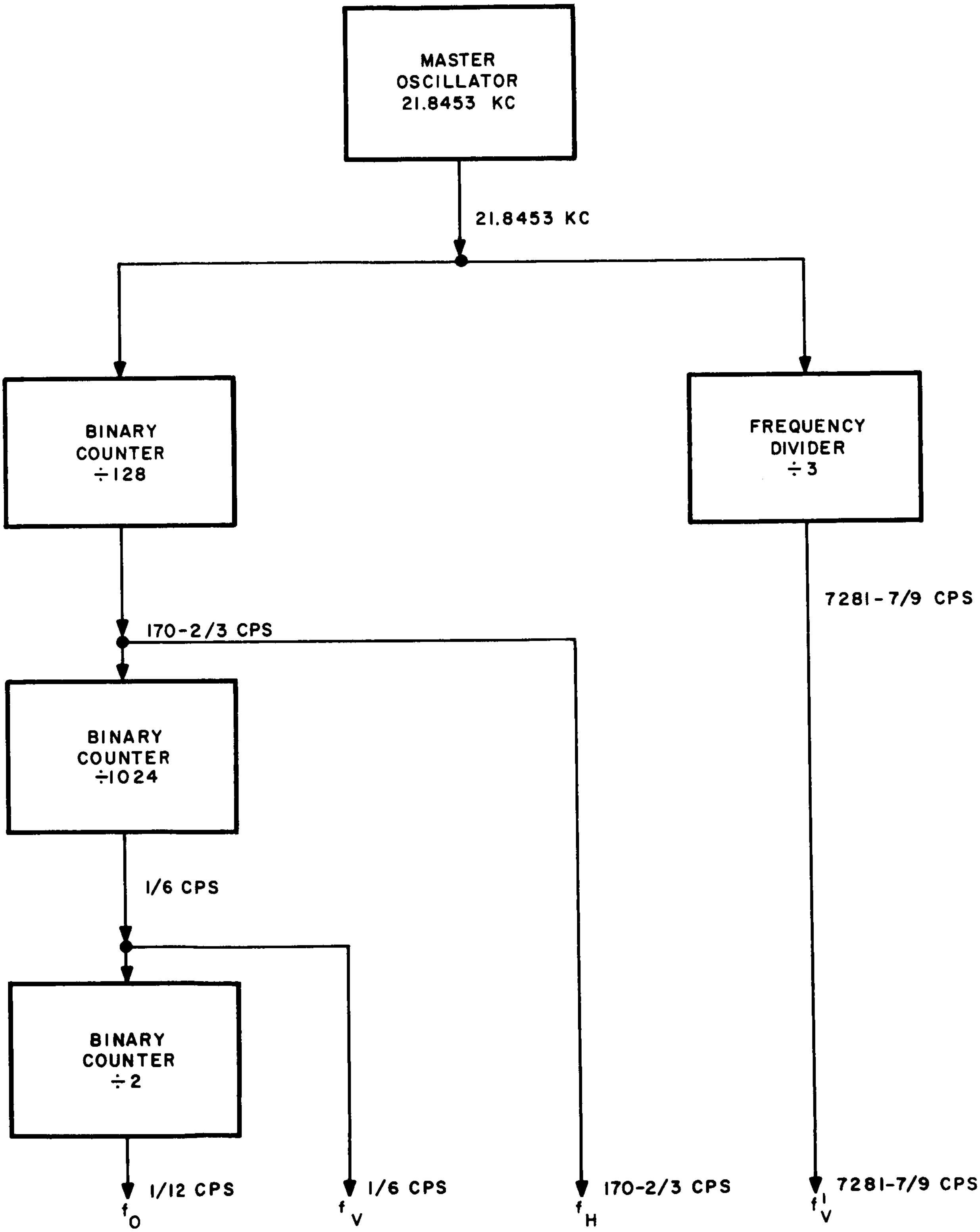


Figure 7. Block Diagram of the Counter Circuits of the Generator

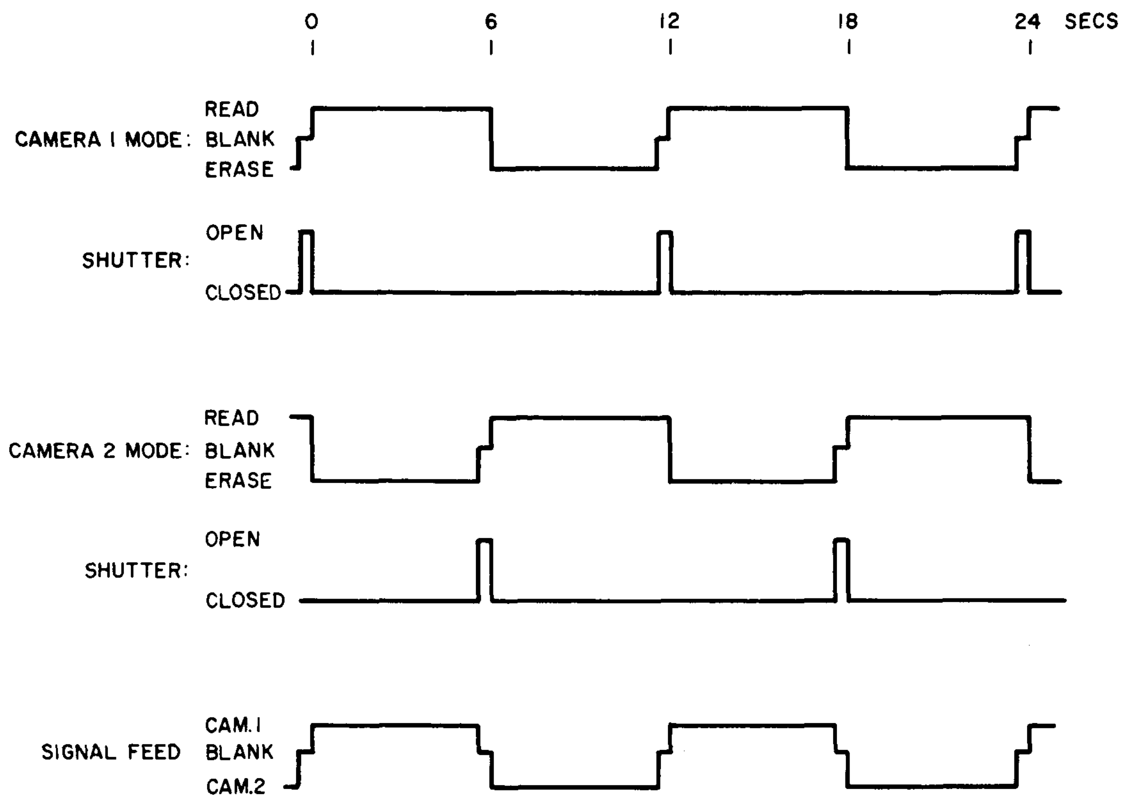


Figure 8. Basic Camera Timing Diagrams

Various other signals are required, but these are formed in the appropriate unit and timed from the signals listed above. For example, the shutter is driven by a shutter drive circuit which is contained in the camera electronic unit and which is in turn driven by the shutter pulse. The required signals for cathode and G-1 of the vidicon are also formed in the camera electronic unit.

D. CAMERA

The camera contains a vidicon camera tube mounted in a focus-deflection yoke assembly. Associated with the vidicon is an electromagnetic focal plane shutter which is timed by means of the shutter pulse to open just before the readout interval (see Figure 8). During readout an appropriate value of beam current is established (with blanking signals for the return beam) by means of suitable signal waveforms applied to the grid and cathode of the vidicon. During erase a uniform illumination is applied to the vidicon target for a period of about 200 msec. This serves to dissipate the charge left on the photoconductor by the preceding exposure. Following this, the beam current is increased and the vertical deflection frequency is changed to the value used for erase (approximately 7000 cps), giving about 1000 frames of erase-scan during the six-second erase interval.

Light input to the vidicon is controlled by an electrically operated iris diaphragm. The iris is actuated by a motor, and its degree of opening is indicated by the voltage at the arm of a potentiometer geared to the iris drive. This voltage is compared with the voltage provided by a fixed voltage divider, and the resulting motion operates to balance the Wheatstone Bridge. Several pre-set voltage dividers are provided, each corresponding to one of a set of desired apertures. The desired aperture is chosen by selecting the appropriate voltage divider by ground command through the command receiver. Iris drive motors are provided in duplicate to improve relibaility.

A video preamplifier is provided to raise the level of the vidicon output, and to provide a suitable drive for the video amplifier contained in the camera electronic unit.

A block diagram of the camera is shown in Figure 9. This illustrates the points covered above, and shows a telemetry output for case temperature.

E. CAMERA ELECTRONICS UNIT

This unit accepts H Sync, V Sync, V' Sync, shutter pulse, erase pulse, and camera gate pulse from the synchronizing generator. Using these signals it develops deflecting current waveforms for both horizontal and vertical deflection and commutates the vertical deflection between that used for read (1/6 cps) and that used for erase (7281-7/9 cps)

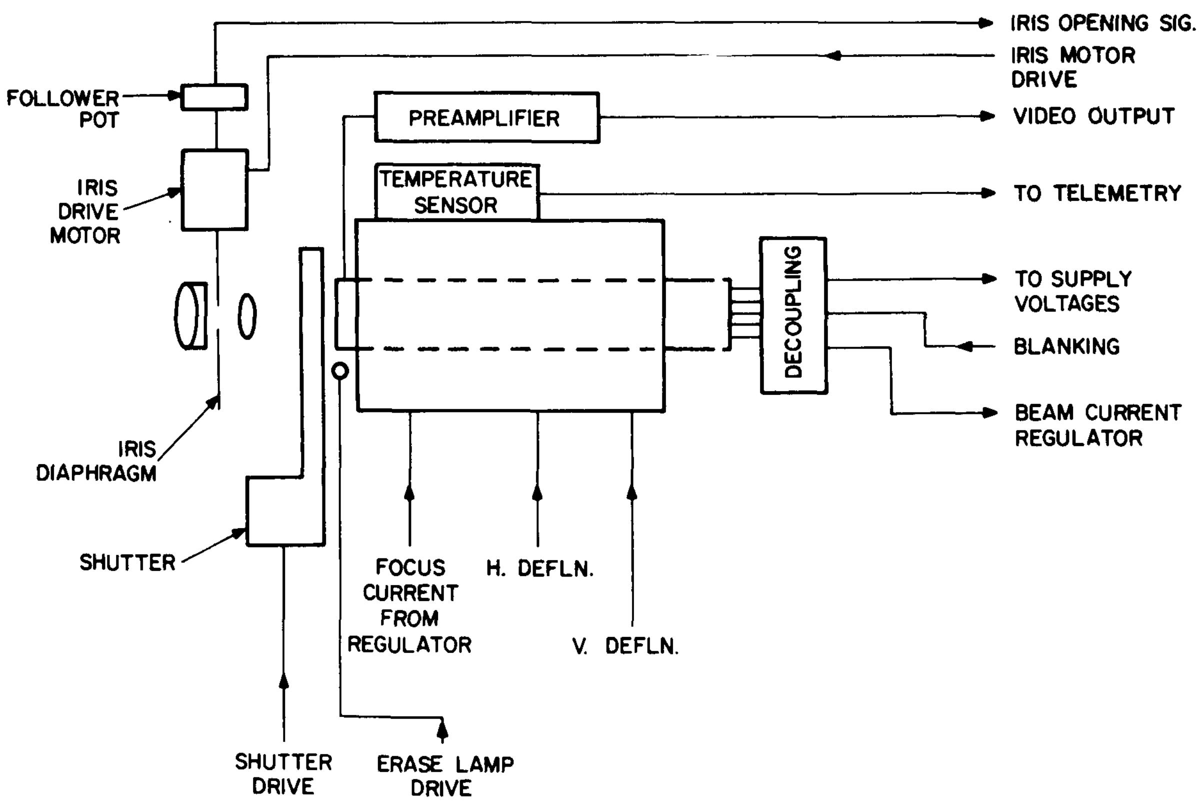


Figure 9. Block Diagram of the TV Camera

by using the camera gating signal. It provides output circuits (which are timed from the shutter pulse) to drive the shutter, and it furnishes output pulses to the erase lamp which are timed from the erase pulse.

The circuit for controlling the iris drive motor to balance the Wheatstone Bridge is part of this unit, and commands select which of the pre-set voltage dividers is to be used in effecting the balance. Power supplies for the camera tube are included, as is the focus current regulation and the beam current regulator. A video amplifier is provided to raise the level provided by the preamplifier in the camera, in order to drive the combiner amplifier.

There are seven command inputs, three for selection of iris, and four to select alternatives between iris drive motors and clutches. There are six telemetry outputs, namely, video, focus current, camera temperature, camera electronic unit temperature, vidicon high voltage, and iris.

F. COMBINER, DISTRIBUTION, AND POWER SUPPLY UNIT

The basic function of this unit is to accept inputs from the two cameras and gate them so their frames occur in sequence. In addition, it inserts horizontal and vertical blanking while clamping each line at black level. This is followed by a DC-coupled clipper which provides a clean base line accurately related to black. This clipping operation removes any spurious signal occuring in the video, by removing any pulse which tends below black. Were this operation not performed there could be interference with picture sync at the receiver. Following this clipping operation the signal passes through an AGC amplifier which is intended to keep its level reasonably constant in order to ensure a constant transmitter modulation, thus avoiding unnecessary addition of noise in the transmission link. Finally, mixed synchronizing is added to the signal. Simple rectangular pulses provide the horizontal sync; but the vertical sync consists of a serrated signal similar to that used in EIA sync, except that equalizing pulses and double frequency serrations are not needed as the system is not interlaced. To retain horizontal sync at the receiver during transmission of the vertical, the leading edges of the vertical blocks are coherent with those of the horizontal pulses.

The unit serves to distribute power to the camera electronics units and the sync generator, and provides the main ON-OFF switch for the television subsystem. It also distributes sync and timing signals to the camera electronics units and provides all telemetry outputs, some of which are generated within the unit. In addition it provides a pair of ±6.3-volt power supplies, which are mutually redundant. It also accepts commands to select between these supplies.

The unit accepts the following commands:

Power Supply A	ON
Power Supply A	OFF
Power Supply B	ON
Power Supply B	OFF
TV Subsystem	ON
TV Subsystem	OFF

It provides the following telemetry outputs which together represent complete telemetry information for the television subsystem:

Vidicon Heater 1	
Vidicon Heater 2	
H Sync	
V Sync	
Video	
Vidicon magnetic focus	
Camera temperature	
Camera electronics temperature	Camera No. 1
Vidicon high voltage	
Iris	
Video	
Vidicon magnetic focus	
Camera temperature	
Camera electronics temperature	Camera No. 2
Vidicon high voltage	
Iris	
Composite video output	

A block diagram of the combiner unit is shown in Figure 10. The outputs of the cameras feed into gates which are controlled by camera gate signals. These signals serve to gate in the video signal while their associated cameras are reading, and to gate out the spurious signals generated during erase. The outputs of the gates are added in a resistive adder circuit and are amplified and clamped. Mixed blanking is then added and the signal is clipped before the DC component (restored by the clamping) is lost. At this point the signal contains an accurate black reference. It is then passed through an AGC amplifier to cause its peak-to-peak amplitude to assume a chosen value. Mixed sync is then added in a sync adding stage.

G. DISCUSSION OF PROBLEM AREAS

1. ERASE

In order to have satisfactory operation of the system as proposed, it will be necessary to provide adequate erasure within a six-second period. Experience indicates that an important feature in obtaining erasure involves making a large number of passes of the scanning beam over the target during the erase cycle. It is for that reason that the erase system described has been provided, since it has proved itself in another camera system.

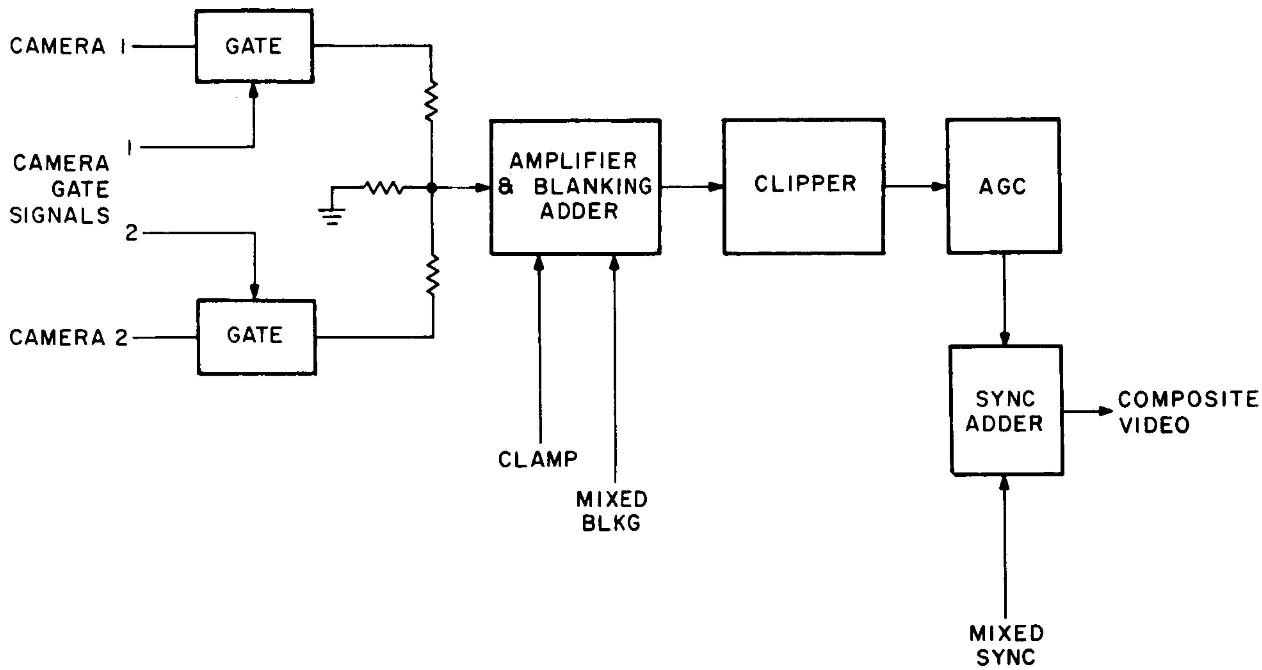


Figure 10. Block Diagram of the Combiner Unit

2. SMEAR

This results from motion of the intersection of the optic axis at the surface being imaged. The motion arises from the velocity of the satellite in combination with its attitude perturbations. Conservative design would indicate that the motion should be small compared with the projection of a scanning line on the lunar surface. However, this places a restraint upon the exposure time, and the higher the resolution the greater is the restraint. And since the shorter the exposure time, the greater must be the image brightness for a given signal-to-noise ratio, it is evident that a compromise must be made in which the relative importance of these factors are considered.

3. SHUTTER SHOCK

The actuation of the electromagnetic shutter causes a mechanical shock to be applied to the camera housing as the shutter armature hits the mechanical stop which limits its travel. This shock can cause microphonics to appear in the video output; at least one source of these microphonics is the camera tube itself. The technique of waiting for the microphonic oscillations to die down before commencing readout is effective but results in reducing the time available for erasure. It is anticipated that some improvement of the present situation will be reqired.

H. ESTIMATES OF SIZE, POWER AND WEIGHT

The estimates shown in Table 7 are based on recorded data from the Nimbus Television System, to which this system bears some similarity. Points of difference include:

(1) The Nimbus system has no synchronizing generator, so some additional weight must be allowed for this.

(2) The Nimbus system employs three cameras, the proposed system has only two.

(3) The erasure required on the proposed system must take place within the time of a single frame, whereas several frames are used in Nimbus.

Table 7 shows the approximate sizes and weights of the units:

TABLE 7. DIMENSION ESTIMATES OF TV SYSTEM COMPONENTS

Unit	Overall Dimensions Inches	Weight lbs.	Number Required	Weight Per System
Camera	8 x 6-1/2 x 7*	7	2	14
Camera Electronics	6 x 13 x 2	5.7	2	11.4
Sync Generator	6 x 13 x 2	4.3	1	4.3
Combiner - Power Supply	6 x 13 x 2	5	1	5
Total Weight not including lenses				34.7

*Not including lenses. Main body roughly conical in shape, 6-1/2" diameter x 7" high. The 8" dimension represents the 1-1/2" protrusion of shutter on one side.

The total power drain is estimated to be in the order of 50 watts total.

SECTION VI

POWER SUPPLY SUBSYSTEM

A. INTRODUCTION

In order to fulfill the power supply requirements for the LOC vehicle, the following possible modes for power systems have been considered:

(1) All-solar cell power supply system
(2) Radioisotope-fuelled thermoelectric generator (RTG) of the SNAP type power supply system

Two possible modes of operation are considered for each of the two basic systems listed above. Power-time profiles for both modes are identical, except for the continuous power levels which are:

27 watts (Mode No. 1)
15 watts (Mode No. 2)

B. SOLAR CELL POWER SUPPLY

The power-time profile for this system is of the worst-case type in that the transmitter peak is said to occur during the shaded portion of flight only. Design is, therefore, based on that profile for both modes of operation. Other assumptions made are the following:

1. battery watt-hour charge-discharge efficiency is 63%.
2. solar cell packing factor is 0.85
3. losses within the array due to off-maximum power-point operation and wiring losses result in a degradation factor of 0.92.
4. regulator electronics power transfer efficiency is 68% during daytime, and 78% during nightime and during large daytime peaks.
5. all loads are to be supplied at a regulated voltage level of -27.5 ±0.5 volts, except the TV transmitter where battery-limited unregulated bus is required (transmitter peak energy is not subject to regulatory electronics losses).

Based on these assumptions, the all-solar cell power supply system was computed and the characteristics of this system are given in Table 8.

TABLE 8 ALL-SOLAR CELL POWER SUPPLY SYSTEM CHARACTERISTICS

	Mode No. 1	Mode No. 2
AE product	1.14	0.845
Area (seq. ft.) E = 0.1	11.4	8.45
Array output req'd (watts)	115	85.8
Array weight (lb.) (incl. structure)	45 ±10%	34 ±10%
Weight of one battery (lb.)	9.6	9.6
Weight of one battery pack (lb.)	12.5	12.5
Total weight (lb.) (no redundancy	70, approx.	60, approx.
Total weight (lb.) (50% redundancy)	82.5, approx.	72.5, approx.
Maximum actual array temp. (^{o}C)	+60	+30

As noted, the AE and area requirements shown in the table are for near-room temperature conditions. In Mode No. 1,this is assumed to be the actual case, whereas in Mode No. 2, the array temperature will be considerably higher. In terms of the overall weight of Mode No. 1, will increase by another 12 to 14 pounds due to array area increase to compensate for the lowered efficiency due to the high temperature.

Redundancy noted in the table is battery-pack redundancy. Each pack weighs 12.5 lbs. maximum with two packs required to maintain operation during the first 27 days in orbit; thereafter, only one pack is sufficient. Redundancy, therefore, is subject to definition in this case. A fully redundant system at all times would require three packs.

C. RTG POWER SUPPLY SYSTEM

The total required power to load for this system is as follows:

Operating Condition	Mode No. 1	Mode No. 2
Sun Time	72.6 w-hr.	38.4 w-hr.
Dark Time	57.5 w-hr.	46.0 w-hr.
Total per orbit	130.1 w-hr.	84.4 w-hr.

The power required from the RTG unit per orbit for Model No. 1 is calculated as follows:

$$(2.25)\ P_{RTG} = 130 \text{ w-hr}$$

$$P_{RTG} = \frac{130 \text{ w-hr}}{2.25 \text{ hr}} = 57.8 \text{ watts}$$

The power required from RTG unit per orbit for Model No. 2 is calculated as follows:

$$(2.25)\ P_{RTG} = 84.4 \text{ w-hr}$$

$$P_{RTG} = \frac{84.4}{2.25} = 37.5 \text{ watts}$$

For peak power supply requirements nickel-codmium batteries will be used.

Table 9 shows the weight characteristics of the 60-watt RTG.

Table 10 shows the performance characteristis of the 60-watt RTG.

TABLE 9 - RADIOISOTOPE-FUELLED THERMOELECTRIC GENERATOR POWER SUPPLY SYSTEM WEIGHT SUMMARY

Item		Weight in lb.
60-watt Generator		50
Storage Battery (including redundancy)		16
Associated Electronics including DC-DC conv.		7
	Total	73 lb
Total weight of the RTG system assuming approximation 5 lb. structure for the main payload enclosure		5
	Total	78 lb

TABLE 10 PERFORMANCE SUMMARY OF THE 60-WATT GENERATOR

Item	Proposed Generator Characteristic
Fuel	Pu-238
Fuel Weight	3290 gms (at beginning of 20 years life)
Element Material:	
P Type	1.0 atomic % Na-doped PbTe
N Type	0.03% Mol PbI Doped PbTe
No. of Couples	140 (capable of growth to 144)
Element Size:	
Length	0.625 in.
P dia.	0.458 in.
N dia.	0.422 in.
Hot Junction Temp.	950°F
Cold Junction Temp.	400°F
Skin Temp.	374°F
Δ T	550°F
Initial Heat Input	1367 watts
Heat Input - after 20 years	1155 watts
Life of the device	20 years
Element Efficiency	5.45%
Generator Efficiency	5.20%
Output :	
Open Circuit Volt.	20.8 volts
Loaded voltage	11.6 volts
Useful Power	60 watts
Overall Weight of the device	50 pounds
Shielding	not required